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SUMMARY TECHNICAL REPORT
OF THE
NATIONAL DEFENSE RESEARCH COMMITTEE

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SUMMARY TECHNICAL REPORT OF DIVISION 3, NDRC

VOLUME 1

ROCKET AND UNDERWATER ORDNANCE

OFFICE OF SCIENTIFIC RESEARCH AND DEVELOPMENT
VANNEVAR BUSH, DIRECTOR

NATIONAL DEFENSE RESEARCH COMMITTEE
JAMES B. CONANT, CHAIRMAN

DIVISION 3
F. L. HOVDE, CHIEF

WASHINGTON, D. C., 1946

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NOTES ON THE ORGANIZATION OF NDRC

The duties of the National Defense Research Committee were (1) to recommend to the Director of OSRD suitable projects and research programs on the instrumentalities of warfare, together with contract facilities for carrying out these projects and programs, and (2) to administer the technical and scientific work of the contracts. More specifically, NDRC functioned by initiating research projects on requests from the Army or the Navy, or on requests from an allied government transmitted through the Liaison Office of OSRD, or on its own considered initiative as a result of the experience of its members. Proposals prepared by the Division, Panel, or Committee for research contracts for performance of the work involved in such projects were first reviewed by NDRC, and if approved, recommended to the Director of OSRD. Upon approval of a proposal by the Director, a contract permitting maximum flexibility of scientific effort was arranged. The business aspects of the contract, including such matters as materials, clearances, vouchers, patents, priorities, legal matters, and administration of patent matters were handled by the Executive Secretary of OSRD.

Originally NDRC administered its work through five divisions, each headed by one of the NDRC members. These were:

- Division A — Armor and Ordnance
- Division B — Bombs, Fuels, Gases, & Chemical Problems
- Division C — Communication and Transportation
- Division D — Detection, Controls, and Instruments
- Division E — Patents and Inventions

In a reorganization in the fall of 1942, twenty-three administrative divisions, panels, or committees were created, each with a chief selected on the basis of his outstanding work in the particular field. The NDRC members then became a reviewing and advisory group to the Director of OSRD. The final organization was as follows:

- Division 1 — Ballistic Research
- Division 2 — Effects of Impact and Explosion
- Division 3 — Rocket Ordnance
- Division 4 — Ordnance Accessories
- Division 5 — New Missiles
- Division 6 — Sub-Surface Warfare
- Division 7 — Fire Control
- Division 8 — Explosives
- Division 9 — Chemistry
- Division 10 — Absorbents and Aerosols
- Division 11 — Chemical Engineering
- Division 12 — Transportation
- Division 13 — Electrical Communication
- Division 14 — Radar
- Division 15 — Radio Coordination
- Division 16 — Optics and Camouflage
- Division 17 — Physics
- Division 18 — War Metallurgy
- Division 19 — Miscellaneous
- Applied Mathematics Panel
- Applied Psychology Panel
- Committee on Propagation
- Tropical Deterioration Administrative Committee

NDRC FOREWORD

As events of the years preceding 1940 revealed more and more clearly the seriousness of the world situation, many scientists in this country came to realize the need of organizing scientific research for service in a national emergency. Recommendations which they made to the White House were given careful and sympathetic attention, and as a result the National Defense Research Committee [NDRC] was formed by Executive Order of the President in the summer of 1940. The members of NDRC, appointed by the President, were instructed to supplement the work of the Army and the Navy in the development of the instrumentalities of war. A year later, upon the establishment of the Office of Scientific Research and Development [OSRD], NDRC became one of its units.

The Summary Technical Report of NDRC is a conscientious effort on the part of NDRC to summarize and evaluate its work and to present it in a useful and permanent form. It comprises some seventy volumes broken into groups corresponding to the NDRC Divisions, Panels, and Committees.

The Summary Technical Report of each Division, Panel, or Committee is an integral survey of the work of that group. The first volume of each group's report contains a summary of the report, stating the problems presented and the philosophy of attacking them, and summarizing the results of the research, development, and training activities undertaken. Some volumes may be "state of the art" treatises covering subjects to which various research groups have contributed information. Others may contain descriptions of devices developed in the laboratories. A master index of all these divisional, panel, and committee reports which together constitute the Summary Technical Report of NDRC is contained in a separate volume, which also includes the index of a microfilm record of pertinent technical laboratory reports and reference material.

Some of the NDRC-sponsored researches which had been declassified by the end of 1945 were of sufficient popular interest that it was found desirable to report them in the form of monographs, such as the series on radar by Division 14 and the monograph on sampling inspection by the Applied Mathematics Panel. Since the material treated in them is not duplicated in the Summary Technical Report of NDRC, the monographs are an important part of the story of these aspects of NDRC research.

In contrast to the information on radar, which is of widespread interest and much of which is released to the public, the research on subsurface warfare is largely classified and is of general interest to a more restricted group. As a consequence, the report of Division 6 is found almost entirely in its Summary Technical Report which runs to over twenty volumes. The extent of the work of a division cannot therefore be judged solely by the number of volumes devoted to it in the Summary Technical Report of NDRC: account must be taken of the monographs and available reports published elsewhere.

The beginning of World War II found the United States with no program for the development of rocket weapons. By the end of the war this country was well in the lead, thanks largely to the efforts of Division 3. As a result of proposals by Dr. C. N. Hickman, NDRC rocket work was initiated in 1940 under Division A, with Richard C. Tolman as chairman. The work was carried forward by Division 3 under two chiefs, John T. Tate in 1943 and Frederick L. Hovde through 1945.

The program, carried out by several contractors with Army and Navy cooperation, produced rockets used effectively by our Infantry, Artillery, Navy, and Air Forces against submarines, ships, tanks, beach defenses, and inland positions. By virtue of their lack of recoil, rockets could be launched from men's shoulders, automotive vehicles, small and large ships, and aircraft. One of the first to go into combat was the bazooka, the Infantry's famed *Panzer* destroyer. In landing operations the Navy used barrage rockets effectively to smother Japanese shore defenses. From one Division 3 contract came also important contributions to the development of torpedoes and depth bombs.

The Division 3 Summary Technical Report, prepared under the direction of the Division Chief and authorized by him for publication, outlines the technical and military knowledge resulting from this program. The performance of Division 3 in discovering and summarizing this information, and, even more, in applying it in timely development of new rocket weapons, deserves our admiration and gratitude.

Vannevar Bush, Director
Office of Scientific Research and Development

J. B. Conant, Chairman
National Defense Research Committee

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FOREWORD

Division 3 directed its operations toward two principal, and conflicting, objectives. The first was to develop rocket ordnance which the Army and Navy could and would use as early as possible in World War II. The second was to provide the new knowledge necessary as a basis for development of improved designs and additional types of rocket weapons during a war of uncertain length. Maintaining the proper balance between these aims as the war progressed was a matter of some difficulty, and was achieved only imperfectly.

Most of the Division 3 rockets were developed to provide our military and naval forces with added fire and bombing power to meet tactical situations for which conventional artillery and bombs were unsuited or not effective. Except for the 1200-pound "Tiny Tim" aircraft rocket, all were under 200 pounds in weight. And none of the artillery type service rockets exceeded 1600 feet per second in velocity. All of them employed grains of solid double-base propellants. None had wings or controls.

Within these general limits, the work of Division 3 embraced research, development, design, experimental and pilot production, and many kinds of testing. Certain studies and developments in underwater ordnance were carried on in close association with the broader activities of Division 6 in this field.

In 1940 neither the Army nor the Navy had any rocket projectiles in service or under development. In the period 1942-1945 many types and sizes of rockets, components, launchers, and related ordnance items developed entirely or in part by Division 3 were used in combat in significant quantities and with substantial effects. Among these were the "bazooka" rocket, the "mousetrap" antisubmarine rocket, several types of rockets used primarily for barrages in landing and field artillery operations, and a variety of rockets for aircraft armament. In addition, the division's laboratories doubled the range of the conventional 4.2-inch mortar through the development of new powder charges; another project involved structural modifications of the Mark 13 aircraft torpedo which increased the overall effectiveness of this important weapon several-fold.

In or on the verge of production when the Japanese surrendered were "superbazooka" rockets, a recoilless 4.2-inch rifle, smokeless rockets for assisting the take-off of airplanes and flying boats, and numerous improved types of rocket ordnance already in service. Among the items in advanced development were water-discriminating fuzes for rockets fired from aircraft against ships, powder-powered launchers for V-1 type flying bombs, powder-pressurized flame throwers, rocket propulsion units for mine field clearance devices, and proximity-fuzed rockets for defense against suicide aircraft attacks.

In connection with these developments the division workers mastered many techniques and amassed much knowledge of rockets and other ordnance. This book provides a partial summary of that knowledge, and a guide to much more of it. Not the least of the division's accomplishments has been the production and wide distribution of a large volume of reports on its work.

Throughout its life the division provided consulting and other technical services to both Army and Navy, not merely on their own developments and those of the division, but also in connection with intelligence covering energy developments. Field technical assistance was provided in the Pacific, in Great Britain, and in France. Of continuing value to the Army and Navy are the personnel and facilities acquired by transfer in the process of demobilizing Division 3. Many of the division's principal operations are continuing under the Navy Bureau of Ordnance.

This book was prepared primarily for the use of military personnel entering on duties involving research and development of rockets and underwater ordnance, technically competent in ordnance engineering, but with limited knowledge of these particular fields. The aims have been to summarize the "state of the art" as it developed during the war, and to indicate some of the directions of future research and development which appeared to be most promising or most necessary. The book serves as an introduction to the numerous final and other technical reports submitted by the several division contractors.

In scope this Summary Technical Report does not cover completely the activities of the division. The book is devoted primarily to basic phenomena, analysis, and methods; the development and design of weapons and other equipment is covered only generally. In Chapters 18, 19, and 20, C. W. Snyder sketches the evolution of most of the rocket designs developed under Section L. It is regrettable that there is no comparable survey of the numerous items developed under Section H; however, complete reports on these have been distributed. Fuzes, launchers, and rocket heads are treated only briefly. Among the subjects not covered at all are production, fire control, terminal ballistics, and tactical employment. Army and Navy experience in rocket development, production, testing, training, and combat employment is not included, except indirectly as it affected the work of the division. The book is historical only where such treatment seemed to its authors to give the most effective exposition.

In Part I, Dr. Max Mason and Dr. F. C. Lindvall summarize the underwater ordnance activities carried out in Division 3 to supplement the broader program of Division 6. Dr. B. H. Sage, in Part II, and Dr. R. E. Gibson, in Part III, treat the problems which lie at the core of rocket development, namely, those of propellants and interior ballistics. C. W. Snyder covers complete rockets, their launchers and their uses in Part IV, and the theories underlying their design and performance in Part V.

Other volumes of the NDRC Summary Technical Report Series include subjects related to the work of Division 3, as follows:

Division 1	Propellants, interior ballistics, gun erosion
Division 2	Terminal ballistics, choice of weapons (including rockets) for specified targets
Division 4	Proximity and other fuzes for rockets, "tossing" of rockets from airplanes
Division 6	Antisubmarine weapons, aircraft torpedoes, hydrodynamics
Division 7	Fire control for rockets
Division 8	Propellants, long-burning rockets, and high explosives
Division 11	Flame throwers and incendiary rockets
Division 12	Use of barrage rockets from DUKW's
Division 14	Radar ranging for aircraft rocket fire control
Division 18	Metallurgy applicable to rockets
Division 19	Rocket armament for guerilla warfare
Applied Mathematics Panel	Theory of heat transfer and of nozzles, analysis of propellant specifications

The NDRC rocket development program was initiated in 1940. Its foundations were laid in Division A under the wise and far-sighted guidance of its Chairman, Dr. Richard C. Tolman, its Vice-Chairman, Dr. Charles C. Lauritsen, the Chairman of its Section H, Dr. Clarence N. Hickman, and, in 1942, the Chairman of its Section C, Dr. John T. Tate. In the NDRC reorganization at the end of 1943 these two sections were merged to form Division 3, with Doctor Tate as Chief. The program continued to grow rapidly. In the summer of 1943 Doctor Tate resigned to devote full time to his responsibilities as Chief of Division 6.

In September 1943, I became Chief of the Division and Acting Chief of its Section L, which was, in effect, a re-established Section C. Section H was reconstituted with Doctor Hickman as Chief. This organization continued through 1945. Principal personnel of these several organizations is shown in an appendix.

The experience of Division 3 demonstrates conclusively that nonmilitary scientists can grasp quickly the needs of the fighting arms and the problems of the supply services, develop new and improved weapons and equipment rapidly, within the limitations of available knowledge, expand that knowledge as required, and on this basis develop still newer and better items. In initiating such a program on the eve of war, the principle of exploring thoroughly, yet quickly, and correlating the technical knowledge available with the apparent operational needs of the war requires no defense. The impor-

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tance of bringing the best scientists into the program as early and in as large numbers as possible has been proved; only thus can effective leadership be provided. Facilities must be provided rapidly, but with a view toward expansion by severalfold. Constant evaluation of promise, progress and results is called for, as a basis for any needed redirection.

It became apparent that the military principle of economy of force applies perhaps more strongly to wartime research and development. This is to say, more valuable results can be achieved sooner by early concentration on those few objectives of greatest value or promising of earliest attainment, to the exclusion, at least temporarily, of perhaps more attractive but less significant objectives. However, small holding and scouting forces are always needed, to consolidate developments and to discover other promising lines of attack. The experience of the division showed the values of follow-through by the applied science forces into the fields of production, testing, training, and analysis of performance under conditions of ultimate service. Another analogy with military operations became apparent, namely, the necessity for prompt and complete abandonment of certain projects as soon as there is a conclusive determination that, in comparison with other projects, the probabilities of early enough success are not in proportion to the effort required. Finally, the experiences of this division and others established new highs in teamwork between military personnel and scientists outside of the military organizations.

Under the present conditions of peace, with time scale and other factors radically changed, research and development operations by or for the services must be governed by principles differing somewhat from those above. I am convinced that the services must continue to have principal responsibility for the development of new weapons and other instrumentalities of warfare. Further, the services must provide for and supervise much more applied research, especially in the fields of their specialized requirements, than heretofore. For many reasons it seems both wise and necessary that they continue strong fundamental research activities in their own military laboratories, yet at the same time promote an extensive and thorough extramural research program in order that the civilian scientists of the nation may continue to serve the needs of national defense in peace as well as in war.

Whatever success the division attained is due in large measure to Dr. Vannevar Bush, Director of the Office of Scientific Research and Development, and to Dr. Irvin Stewart, Executive Secretary and Contracting Officer, and their staffs. Under their wise policies, flexible organizations and effective operating procedures, a majority of the nation's scientists and scientific organizations performed an unprecedented job with a degree of efficiency and coordination unusual in government operations in war or in peace. A basic element was the freedom allowed the divisions and contractors in choosing and using various means for achieving approved objectives. Dr. James B. Conant, Chairman, and the members of the National Defense Research Committee, with their staffs, were responsible for approving the proposals of Division 3, and for reviewing and coordinating its work with that of other divisions.

To the British government and to British scientists we owe a tremendous debt for making freely available their knowledge and experience gained in several years of defense research prior to the advent of NDRC and in active warfare preceding that of the United States. On the OSRD Liaison Office fell the burden of arranging for and handling this international exchange of information and of scientific personnel. This exchange, especially in the early years, made possible a manifold increase in the division's rate of progress.

Liaison organizations and offices of the War and Navy Departments, and their cooperating field units, provided guidance as to specific service needs, participated in some phases of Division 3 developments, and expedited their transitions to combat employment.

Many other NDRC divisions made available knowledge and services to hasten Division 3 work, and included in their programs complementary projects which increased the utility of Division 3 developments to the Armed Forces.

The functions of initiating, establishing, guiding, supervising, and administering the operations of Division 3 were well performed by its highly competent members, consultants and staff, and by the able staffs of the two sections. I am deeply grateful to all of them for faithful and talented services and for the privilege of working with them.

The principal credit, of course, must go to the several contracting organizations (listed in an appendix) under which all of the Division 3 research and development was carried out. To them, and even more to their personnel, who furnished the ideas, knowledge, skills, and plain hard work which constituted the program, is due whatever praise the division may have earned.

In conclusion, I express my appreciation to the six authors who contributed to this Summary Technical Report. For it they gave of their time, talents, and efforts in the face of the pressing demands of their postwar activities, with little indication that the results would be worth the effort. As for myself, I am confident that they have produced a volume which will provide proper perspective for the numerous reports of the division, and which will, in conjunction with those reports, preserve most of the benefits of the division's five years of wartime ordnance development.

FREDERICK L. HOVDE
Chief, Division 3

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PREFACE

THE GENERAL SCOPE and results of the Division 3 program are indicated in the Foreword by Frederick L. Hovde. The activities of the Division involved the services of approximately 800 scientists and engineers working under eleven prime contracts during the period 1940-1945. Total costs were of the order of $25,000,000 for research and development and $50,000,000 for experimental and pilot production.

As a part of the effort to preserve the values of the Division's work, this summary technical report was prepared, primarily for the orientation of technical officers, engineers, and scientists who seek to acquire familiarity with the basic phenomena of solid fuel rockets or of the entrance of underwater ordnance into water. The volume may be useful also to more experienced workers in these fields, for review or reference purposes. The principles and important results of the Division program are summarized as of the end of 1945, as a foundation for the study of the substantial advances made thereafter by others.

In this summary, the treatment of the subjects listed in the Contents, though it is technical, does not require previous knowledge of the subjects. Throughout the book, the emphasis is on technical considerations pertinent to military applications. Chapter 14 includes analyses of the military utility of solid fuel rockets.

The information in this report is arranged in five parts by authors and subjects, rather than by projects. Each chapter was written by a single author who led Division 3 developments in the fields which he treats. The four authors of Parts I, II, IV, and V were associated with the single Section L contract, number OEMsr-418 with the California Institute of Technology. The two authors of Part III were concerned with the activities under all ten Section H contracts; they were affiliated with the Allegany Ballistics Laboratory, which was operated by the George Washington University. The fact that each of the six authors has written mainly on the experience in his organization, and in a manner of his own choosing, has resulted in a division of the text of this report on the basis of the sections and contracts indicated.

As a result of this situation, the very important subjects of propellants and interior ballistics are presented from three points of view. In Part II Dr. Sage analyzes the problems of developing, designing, and producing rocket propellant charges of compositions of the sort employed in all United States rockets which saw combat in World War II. These compositions are generally similar to that of trench mortar sheet powder. In Part V, C. W. Snyder reviews these problems from the viewpoint of the projectile designer. Dr. Gibson and Dr. McClure describe in Part III the behavior of solid propellants of a much broader range of chemical composition.

The functions of the volume technical editor have varied for different parts of the report, but in general they have been limited to minor revisions and rearrangements of the authors' material, and the addition of somewhat inadequate footnotes, most of them referring to related subject coverage by the other authors.

Mathematical treatments have been limited to relationships of fundamental importance, with details of their derivation and application covered only by references to other reports. The mathematical symbols are consistent for each author but not entirely uniform among them. Most of the symbols are the same as those used in reports previously issued by the authors' organizations.

Because of the pressure of more urgent work, it was not possible to start the writing of this summary technical report before the surrender of Japan. After that, progress on it was delayed by the discharge of the authors' responsibilities in connection with final reports, contract terminations, transfer of many activities and facilities to the Services, and postwar engagements. Under these and other difficulties the six authors labored manfully to produce the following report. It is the editor's opinion that the advantages derived from their superior qualifications in the subjects covered have amply justified the acceptance of the delays. The authors and the editor have reviewed the galley proofs, but the tight publication schedule has precluded this process on the page proofs.

This volume is a somewhat incomplete summary of the scientific and technological results of Division 3 work. It was not possible, unfortunately, to include much information on the rocket projectiles developed under Section H, or on the nu-

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merous applications of rocket technology by that section to the development of rocket thrust units for airplanes and anti-mine devices, of recoilless guns, and of devices which utilized the burning of rocket propellants as sources of high pressure gases for several purposes. This report outlines the basic principles. For complete information on these and other Division 3 developments, the reader is referred to the General Bibliography appended, in which are listed several hundred of the more important reports of the Division.

In keeping with its character as a technical summary, this report includes information on Division 3 personnel, organization, contracts, and projects only as listings in appendices. No attempt has been made to present the history of rockets or of the Division's work on them, or to describe the combat or other Service experience with Division 3 developments.

A popular account along these lines is available from the Superintendent of Documents under the title "Rocket Ordnance—Development and Use in World War II." Little, Brown and Company have published a series of volumes on OSRD and its contributions to World War II. Of these, the one by Dr. James P. Baxter 3rd is the short history of OSRD. Of the other long history volumes, about half of the one edited by Professor John E. Burchard is a history of Division 3 work, another by Burchard and Thiesmeyer describes the work of OSRD scientists, including several from Division 3, in combat areas, and another, by Dr. Irvin Stewart, outlines the organization and administration of OSRD.

For many reasons, this report has excluded acknowledgments of credit for technical or other contributions to the advancement of the Division program. The titles of reports listed in the appended General Bibliography provide some indications as to the types of contributions made by their authors. The work of the Division was aided greatly by lessons learned from the experience of United States and British Service and civilian agencies in the development, production, testing, and training and combat use of rockets and other ordnance.

The editor acknowledges his gratitude to all the authors for the cooperation they provided under difficult conditions in the preparation of this report. It is hoped that the readers will find enough value in their chapters to justify a generous tolerance of editorial defects. Dr. Gibson, Dr. McClure, and the editor join in acknowledging the helpful review and comment on Part III provided by Dr. Alexander Kossiakoff, former Deputy Director of the Allegany Ballistics Laboratory. Taking advantage of this opportunity, the editor records here the great satisfaction he has derived from several years of pleasant associations with the personnel of OSRD, NDRC, and many of their contracting organizations, and in particular with Dr. Richard C. Tolman, Dr. John T. Tate, and Frederick L. Hovde.

Eliot B. Bradford
Editor

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PART V

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by *E. B. Bradford*

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SUMMARY

by *E. B. Bradford*

Underwater Ordnance

Part I of this report describes briefly the unprecedented facilities developed at Morris Dam (near Pasadena) for full-scale studies of the behavior of aircraft torpedoes and other underwater ordnance items on entry into water at extreme speeds and angles. With these and other facilities, important contributions were made to several of the weapons of World War II, and to better understanding of the phenomena of water entry and underwater travel. The Navy continued these operations after the war. Highlights of the wartime work are summarized in the Introduction to Part I.

Solid Fuel Rockets

Parts II–V summarize most of the principles and practices employed by Division 3 in the development of nearly all the rockets used by United States forces in World War II combat, and of several others not so used. In all these rockets smokeless powders were used. By the end of the war, several types of rockets had demonstrated their utility in many tactical situations, and Navy procurement of them was on a financial scale comparable with conventional ammunition.

Rocket Characteristics and Uses

In nearly all their uses, rockets performed the function of artillery. Lethal or other payloads up to 500 lb were delivered to ranges up to 10,000 yd, with detonation or other effects. By virtue of their self-contained recoil-less propulsion, and the light, simple launchers thus made possible, rockets achieved big-gun effects from such relatively frail mounts as airplanes, small boats, light land vehicles, and men's shoulders. Fired forward from airplanes, fin-stabilized rockets in calibers up to 12 in. were especially useful against small hard targets. For faster airplanes, spin-stabilized rockets offer certain advantages. Rockets used from surface ships included the "mousetrap" antisubmarine depth bomb, several types (finners and spinners) for offshore barrages, and fast spinners (1,540 ft/sec) as main batteries for PT boats. In ground warfare, rocket launchers mounted on trucks and tanks drenched area targets at critical periods.

The launcher plays no part in propulsion and is subjected to little or no recoil force. Its function is simply to guide the initial motion of the rocket along the line of proper train and elevation. This is accomplished by light rails, tubes or slots, or, on airplanes, by the airstream.

On the other side of the picture it must be noted that rockets have disadvantages which may include rearward blast, smoke, flash, lack of accuracy, limited velocity and range, low percentage of weight effective at the target, and variation of performance and safety with temperature.

Rocket Heads, Fuzes, and Effects

The effects achieved at the target by most rockets are those of artillery and aerial bombs. In elementary rocket theory the head is the first item selected or designed, on the basis of target effects desired. Since the accelerations and stresses of projection are low, the problems of head design are generally similar to those of bomb design.

An advantageous property of long-finned rockets is their long straight underwater travel. This characteristic was improved, by blunting the nose curvature, so that 3.5-in. aircraft rockets with solid heads were enabled to perforate submarines after 130 ft of underwater travel, thus making range estimation less critical.

The requirements of function and safety for rocket fuzes are the same as those for shell and bombs. Shell fuzes were adapted for spin-stabilized rockets. For fin-stabilized rounds, with no spin and low setback, the fuzes involved various combinations of mortar fuze adaptations, setback devices, arming wires, air-driven propellers, and time delay. Impact was usually used to trigger detonation, in some cases with time delay.

An extensive series of fuzes was developed, of which many were standardized. One of the last fuze developments provides radically new performance, especially for underwater hits on floating targets. This deceleration discriminating fuze arms partially on first impact with water or target but fires only after it has penetrated the hull (high deceleration) and emerged inside (low deceleration) or after its velocity has dropped to a low value.

Exterior Ballistics

The behavior of rockets in flight and the methods used for its analysis have many similarities to those of shell and bombs. The outstanding differences are due to the continuation of propulsion and acceleration over distances as much as 1,000 ft beyond the launcher. With spin-stabilized rockets the rate of spin continues to increase throughout the period of propulsion. Most of the dispersion of rockets has its origin in this period.

Accuracy has been improved, and the factors affecting it have become better understood, as a result of thorough analyses of the oscillations, precessions, and nutations of rockets in flight. The flight behavior and especially the accuracy of World War II rockets were undesirably sensitive to changes in temperature. As indicated below, propellant developments late in the war improved this situation. Wind is a factor with several effects on rocket flight, some of them related to temperature and all of them tending to reduce accuracy.

Fin stabilization provides simplicity, economy, flexibility in design, and possibilities for various combinations of a few heads and motors to serve many purposes. Spin stabilization has advantages in better accuracy, shorter launchers, easier handling and better adaptability to automatic launchers, but it introduces severe design restrictions. The requirements for flight stability involve relationships among velocity, rate of spin, propellant strength, ratio of length to caliber (commonly 6 to 7) and weight distribution. One result is that different types of use usually require different designs.

Rocket Motors

The function of a rocket motor is to provide an impulse for the acceleration of a projectile or other load. This total impulse is the product of the thrust and its duration, usually expressed in pounds-seconds. The rocket motor produces the thrust as a reaction to its rapid rearward discharge of a stream of gases. In the case of free flight, the impulse given to the whole rocket is equal to the momentum (mass $\times$ velocity) imparted to it, which is equal and opposite to the momentum given the gases.

For each size and type of rocket there is an upper limit to the velocity obtainable, even with the payload reduced to zero. This limit can be raised by increasing the impulse-to-weight ratio of the motor, the motor specific impulse, commonly expressed in pounds-seconds thrust per pound of initial weight of the loaded motor. This ratio is increased by designing for combustion at constant, low pressure in a chamber of high strength-to-weight ratio. A basic requirement is a propellant composition which, burned in a suitably designed motor, gives a high specific impulse. A value typical of World War II rocket propellants is 200 lb-sec thrust per pound of propellant burned. Multiplication of specific impulse by the acceleration of gravity gives the effective gas velocity, frequently used to indicate the performance of a propellant in a rocket. The velocity acquired by the rocket is roughly this effective gas velocity multiplied by the ratio of propellant weight to total weight.

The typical solid fuel rocket motor is a steel tube, closed at the front end, with one or several venturi nozzles at the rear. The nozzles serve to maintain the desired combustion pressure, to smooth and direct the discharge of propellant gases, and, by expanding them, to add about 30 per cent of the total thrust. Motors for finners are usually long and slim, for reasons of aerodynamics, accuracy and economy; spinner motors are rather short, as required for flight stability. Spin is produced by multiple nozzles mounted on a circle at angles resulting in a peripheral component of thrust.

Charge Design

Within the motor is the propellant charge, of weight given by dividing the specific impulse characteristic of its composition into the total impulse required. Constant pressure operation of the rocket motor requires a constant mass rate of discharge of propellant equalled by a constant mass rate of burning, the latter involving parallel layer burning over a constant total surface which recedes at a constant linear rate of burning. Constant burning area may be secured simply by grain shape, or it may involve "inhibiting" certain surfaces to prevent their burning. High density of loading is sought; this leads frequently to a single grain charge. Other considerations may require a multi-grain charge. Low operating pressure is secured by a wide nozzle opening, a small burning area, and a propellant composition of slow linear burning rate.

Characteristics of Solid Propellants

Of fundamental importance in the interior ballistics of rockets are the linear burning rate of the propellant and the increase of this rate with pressure and with temperature. For the propellants used in the rockets which saw combat, the pressure sensitivity was such that the equilibrium motor pressure varied approximately as the fourth power of several motor parameters; newer propellants brought this power down to about 1.2.

The temperature range within which the best World War II rockets gave safe and dependable performance was -40 F to $+140$ F. Pressure, thrust, acceleration, burning time, burning distance, and dispersion varied by factors as high as three between the upper and lower limits, mainly because of the sensitivity of the burning rate to propellant temperature. Propellants developed during the war had temperature coefficients from 1.5 down to 0.1 per cent change in equilibrium motor pressure per degree centigrade.

The improvements in propellant characteristics resulted from studied changes in chemical composition. The physical properties of propellants, especially mechanical toughness, are important to proper performance under the stresses of rocket acceleration. The compositions and characteristics of solid rocket propellants are surveyed in this report, as are processes for propellant production.

Conclusion

Many possibilities for rockets substantially better than those of World War II have been demonstrated; others are indicated. Several chapters of this report include recommendations as to promising lines for future research and development.

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PART I

UNDERWATER ORDNANCE

By *E. B. Bradford* [a]

In its development of rocket ordnance Division 3 and its predecessor units of NDRC led the way in virgin territory; in 1940 neither the Army nor Navy had any activities or much interest in this field. In underwater ordnance, on the other hand, the Services, especially the Navy, had extensive experience and activity. Nevertheless, the civilian and largely academic scientists of NDRC were able to grasp the outstanding problems and contribute effectively to many of them, in the improvement of old weapons like torpedoes, in the development of new ones like ahead-thrown depth bombs, and in the general advance of underwater ordnance research, development, and testing.

In NDRC, Division 6 (formerly Section C4) pursued rather broad programs on underwater ordnance.[b] Certain specialized work in this field was, however, carried out in Division 3, in substantial part for Division 6. All this Division 3 work was done by two special sections of the rocket contract (OEMsr-418) with the California Institute of Technology [CIT]. Section IV was concerned mainly with water entry and underwater performance characteristics of depth bombs, depth charges, and similar ordnance; its activities included full-scale testing of service and experimental ordnance items, model scale studies, and associated theoretical research. Section VII was concerned entirely with aircraft torpedoes, primarily with the fundamental study of the behavior of torpedoes and their components on high-speed entry into water in full-scale tests.

Although both groups had as their prime function the providing of test data and other information for application elsewhere to problems of ordnance design, both participated directly in certain weapon developments which found significant service uses. Among those involving Section IV were ahead-thrown depth bombs of both the spigot-projected (Hedgehog) and rocket-propelled (Mousetrap) types, retro rocket depth bombs for the attack of submarines by MAD-equipped airplanes, and the forward-firing aircraft rockets which were so effective against underwater targets as well as others. Section VII, starting with model test indications from a Division 6 program, developed the shroud ring modification for the tail of the Mk 13 torpedo, and, in the summer of 1944, provided the first 1,000 of these to go to combat areas. This modification eliminated the serious restrictions imposed on pilots by the older torpedoes; with the new ones they were enabled to release their torpedoes at any speeds of which their airplanes were capable, from higher altitudes, and still secure more hot, straight runs than they had formerly from lower and slower approaches, with their greater exposure to AA fire.

In Chapter 1 of this Division 3 Summary Technical Report, Dr. Max Mason, who headed Section IV, outlines its principal activities and results. In Chapters 2, 3, and 4, Dr. F. C. Lindvall summarizes the Section VII work under his supervision. Both of these summaries indicate the scopes of the programs and of the special facilities and instrumentation developed for them. Each serves as an introduction to an OEMsr-418 final report volume (cited) on the work. Several hundred copies of each of these volumes have been distributed through the War and Navy Departments.

Sections IV and VII were both set up initially to provide and operate new and unprecedented facilities for the securing of full-scale test data not obtainable as accurately or as economically by existing practices. The principal facilities of both groups are located at the Morris Dam Reservoir in Southern California. Together with records and experienced personnel, they were taken over by the Navy in late 1945. They are now being expanded and operated under the Naval Ordnance Test Station, Inyokern, California, as parts of the Navy's peacetime underwater ordnance program. The Section IV facilities were designed and used to produce data with laboratory precision from full-scale launchings duplicating pertinent conditions of operational use of several types of underwater ordnance. The data obtained covered air-water trajectories, accuracy, sinking speed, and fuze functioning, as

[a] Volume editor.

[b] See Division 6 Summary Technical Report.

well as the effects of shape and weight distribution on these aspects of performance. The Section VII facilities provided for the launching of torpedoes into water at extreme speeds and angles; rather elaborate external and internal instrumentation was employed to provide detailed information on the behavior of torpedoes and their components under these conditions. Thus, in both cases, it was possible to get more, and more accurate, information than that obtainable from service-type tests, with their complications as to time, weather, manpower, availability and limitations of airplanes, ships, equipment, etc. The method previously used for securing comparable data on torpedoes, for example, had been to drop them from available airplanes (frequently not fast enough) and try to see what happened—the limitations are obvious. With the new facilities, many features of underwater ordnance designs could be established more definitely at earlier stages of development, with service-type testing required for little more than final proof.

In both sections programs of basic research were carried on in association with the testing activities, to provide foundations for further advances in underwater ordnance. These programs are outlined by Mason and Lindvall, and presented in detail in the CIT final reports which they cite as bibliographic references.

To complete the picture of Division 3 torpedo work, an early, stopgap development may be mentioned briefly. In 1943, in an effort to provide a way around the limitations of the Mk 13 torpedo, CIT developed a device which decelerated it by 100 knots between release and water entry. This was accomplished by an assemblage of rocket motors so mounted on the torpedo as to exert rearward thrust during the free fall, and to detach itself before entry. Such devices performed successfully in torpedo-dropping tests at the San Diego Naval Air Station and the Newport Naval Torpedo Station, but were not adopted for service.

In considering the summaries by Mason and Lindvall, it must be remembered that World War II ended with the various research programs in widely differing stages of completion. Hence, although much has been learned about some items, there are many others in which the surface had barely been scratched by the time the activities under the OSRD contract were taken over by the Navy. In these cases the results should be considered as preliminary surveys indicative of the direction in which further work might fruitfully be pursued.

Chapter 1

ANTISUBMARINE WEAPONS AND UNDERWATER BALLISTICS

By *Max Mason* [a]

1.1 INTRODUCTION

The underwater ordnance studies of Section IV of the organization which grew up at the California Institute of Technology under Contract OEMsr-418 had two main aspects: (1) the building up of special facilities at Morris Dam, and their use in tests and development of antisubmarine ordnance, and (2) mathematical and model scale studies of the fundamental ballistics of water entry and underwater travel. These are covered under separate headings in this chapter.

1.2 FULL-SCALE WEAPON TESTING AND DEVELOPMENT

Throughout World War II Morris Dam conducted full-scale and large-model tests for which no comparable facilities were available elsewhere in this country. As a part of the testing program about fifty different service devices of the United States and British Navies were studied, and measurements of their underwater performance reported [b] for evaluation, guidance of design changes, and other uses.

Similar testing services were provided for Division 6 (formerly Section C4). Among the ordnance items to which Morris Dam contributed in this way were the following:

Depth charges, Mks VI, IX, XII, and XVII.
U. S. versions of the British Hedgehog projectile.
The Mk 24 mine.
The antisubmarine scatter bomb of Divisions 3 and 6.
The British Projectile Type C (Squid).
Numerous pistols and fuzes for these and other weapons.

In addition to providing these test services, the Morris Dam group participated directly in the development of several types of rocket ordnance for the attack of underwater targets, as indicated below.

1.2.1 The Problem of Antisubmarine Ordnance

In the period following the first World War the detection and location of submerged submarines by echo ranging ("sonar") was highly developed. By this means both direction and range of a submarine could be determined from a single ship. The standard depth charge remained, however, the only ordnance for attack. This was a very ineffective weapon. Among its shortcomings were slow sinking speed and rather erratic underwater trajectories. Although such depth charges could be thrown from large ships, they had to be dropped from small ones. In both cases the number which could be launched from one ship simultaneously or in quick succession was limited. Their fuzes functioned at preset depths, whether near the submarine or not. Sound contact with the submarine was frequently lost because of the maneuvering required for dropping the depth charges and the disturbances caused by their explosions. Better antisubmarine ordnance, preferably usable from small ships, was urgently needed. This view was emphasized by the results of British statistical studies of depth charge attacks.

Attention was therefore directed to fast-sinking bombs fuzed to detonate only on contact with the submarine, and to the projection of a number of such bombs forward from the hunting ship during a sonar fix. In this way cat-and-mouse tactics could replace the blind-man's-buff method of the depth charge. The effectiveness of this type of antisubmarine armament was indicated by British work on the development of the "Hedgehog," first of the "ahead-thrown" weapons. This consisted of an

[a] Supervisor of Section IV (Underwater Properties of Projectiles) of Contract OEMsr-418 at the California Institute of Technology.

[b] All reports issued by Section IV are included in the general bibliography appended to this volume, under OEMsr-418 file series IBC, IEC, IHC, IIC, IOC, IPC, JHC, and JPC. The bibliography of *Water Entry and Underwater Ballistics of Projectiles*[1] lists these reports under several subject headings. They are listed also by a different subject classification in the NDRC Summary Technical Report Microfilm Index.

array of spigot launchers, from which a substantial number of contact-fuzed depth bombs, each containing about 35 lb of high explosive, were projected over the bow. In the absence of a hit, there was no explosion, and sound contact was retained.

1.2.2 Establishment of Morris Dam Laboratory

Development of "ahead-thrown" ordnance for U. S. production and use required facilities for studying behavior of the projectiles on entering into and proceeding under water, and for observing fuze action. To meet these and related needs, the Morris Dam Laboratory was established in August 1941, by the California Institute of Technology under Contract OEMsr-329 with the Office of Scientific Research and Development; technical supervision for the government was the responsibility of Section C4 (later Division 6) of NDRC. The engineering talent for design and operation of the new facilities came mainly from the CIT 200-in. telescope project, on which activity was suspended during World War II. A general view of the installation is shown in Figure 1. The first work at the Morris Dam was in cooperation with other C4 activities at New London, on the testing of depth charges and the design of fast-sinking bombs. Studies of rocket-propelled antisubmarine ordnance soon became an important activity, and from June of 1942 the activities were included in those under Contract OEMsr-418, which covered the CIT rocket developments then under Section C of Division A, NDRC, and, after December 1942, under Division 3.

FIGURE 1. General view of Morris Dam and testing facilities. Splash near the center of the picture indicates projectile has just been launched down one of the ramps. Nets and targets used for determining trajectories and recovering the projectiles are shown at the right.

[redacted]

1.2.3 Mousetrap—an "Ahead-Thrown" Depth Bomb without Recoil[c]

Because of its recoil effects, the Hedgehog was usable only on fairly large ships, with well-braced foredecks. In 1942 there were not available enough such craft to meet the urgent submarine situation. To provide equivalent striking power for smaller craft, CIT developed a weapon similar in use and effectiveness to the Hedgehog, but with recoilless rocket projection instead of spigot gun projection.

This armament, known as "Mousetrap," resulted from collaboration of the Morris Dam group with the rocket group. Its development involved determination of the best head shape, weight distribution, and fin configuration to provide maximum accuracy in launching, air flight, oblique water entry, and sinking. With this weapon many smaller ships were equipped with effectively the same attack power as destroyers, and antisubmarine patrols were substantially strengthened.

1.2.4 Retro Bombs for Antisubmarine Aircraft[c]

The development of the *magnetic airborne detector* [MAD] presented an analogous problem. Until the advent of the sonobuoy, MAD was the only device by which an airplane could detect an invisible, submerged submarine. However, it indicated location only when directly over the submarine. Conventional aircraft armament was at a disadvantage in this situation. The rocket and underwater ordnance groups at CIT collaborated again, to conceive and develop a type of armament suitable for use with MAD. For the ammunition, heads adapted from the Mousetrap were used, mounted on rocket motors which propelled them at speeds to match aircraft cruising speeds. Usually mounted twelve under each wing, these were fired backward on MAD indications (after exploratory passes) to enter the water in a pattern across the area in which the submarine had been located. Here the problem was one of substantially vertical fall, water entry, and sinking, with the accuracy problem complicated by oscillation of the missiles at entry.

[c] The Mousetrap rockets are described briefly in Chapter 18; retro-rockets and their components, launchers and employment are covered at greater length in Bureau of Ordnance publications and other reports listed in the general bibliography appended to this volume.

1.2.5 Aircraft Rockets for Underwater Targets[c]

The third and most successful project on which the Morris Dam group collaborated with the Division 3 rocket workers at CIT was the development of rockets which, fired forward from diving aircraft to enter the water at high speed, and after some distance of underwater travel, would hit an underwater target with energy enough to penetrate the hulls of submarines and thin-skinned ships. Here again the ballistics of air flight, water entry, and underwater travel had to be combined to secure maximum range and accuracy and to determine the best dive angles (and hence water entry angles) for attacks.

1.2.6 Facilities for Testing Underwater Performance

A major part of the work of the Morris Dam group was the devising of instrumental means of study and measurement. The principal facilities, described in detail in the Section IV final report,[1] are summarized in the following paragraphs. Except for item 6, all these are at Morris Dam.

1. A large sound range for observing time-position relations, with a horizontal recovery target 50 ft by 50 ft which can be lowered to 180-ft depth of water and a vertical target 62 ft by 70 ft for shallow entry. These can be seen in Figure 1. Continuous records are obtained from six hydrophones and a six-channel oscillograph. The coordinates of underwater trajectories are obtained without arithmetic reckoning by a special computing device.

2. An electrical net, and other net equipment, for determining shallow trajectories which cannot be evaluated with sufficient precision by the sound range.

3. Rocket and blowgun launching facilities, adjustable for obtaining desired air trajectories or entry angles, with entry velocities as high as 1,000 fps for 1-in. diameter specimens and about 900 fps with 70-lb projectiles. The high entry velocities are a special objective of this facility.

4. Facilities for taking underwater motion pictures of bubble and cavitation phenomena down to the maximum depth of the lake.

5. Facilities for underwater impact tests and fuze tests.

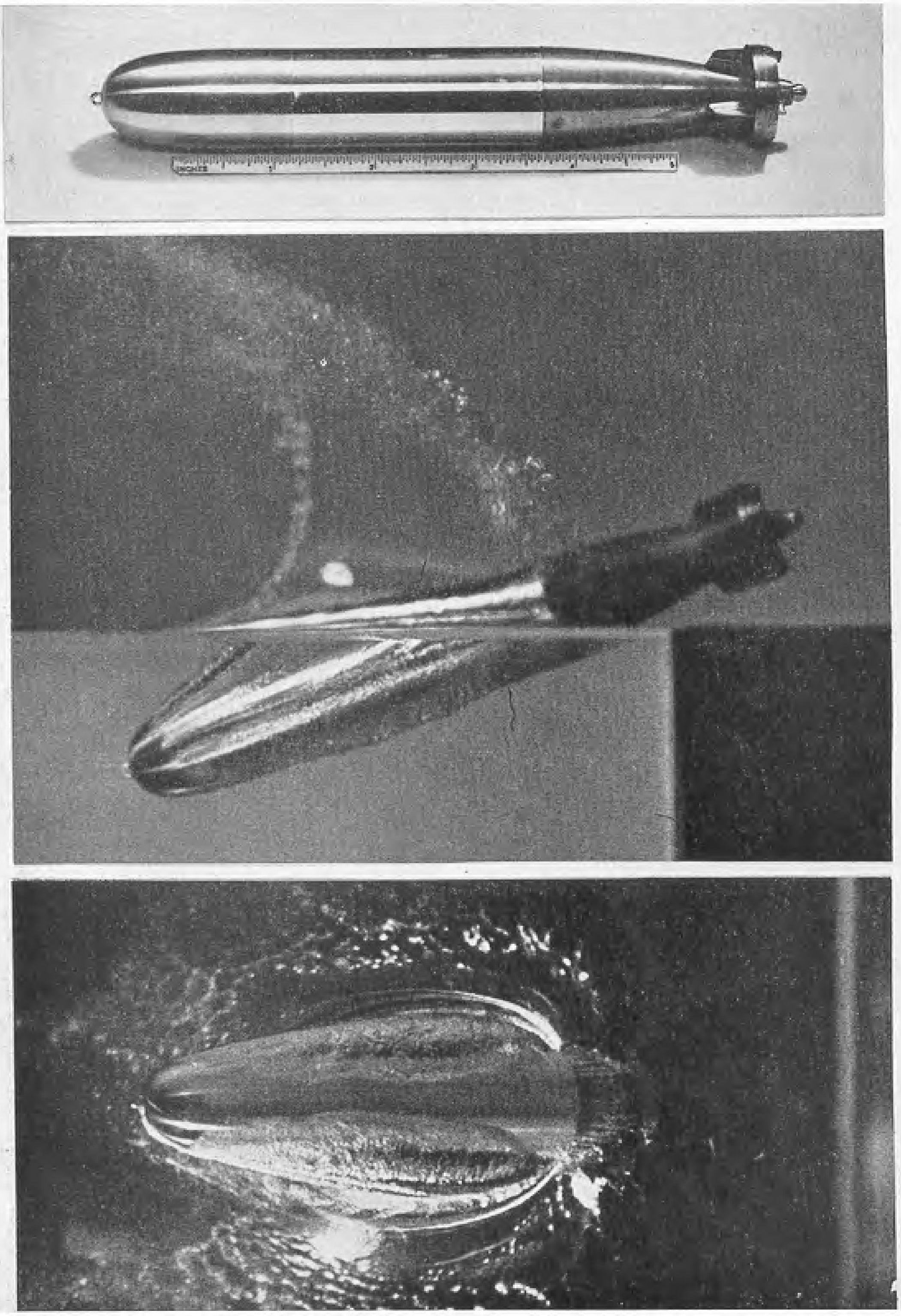

FIGURE 2. Behavior of unvented torpedo model. Side and bottom views of oblique entry show how the water clings to underside.

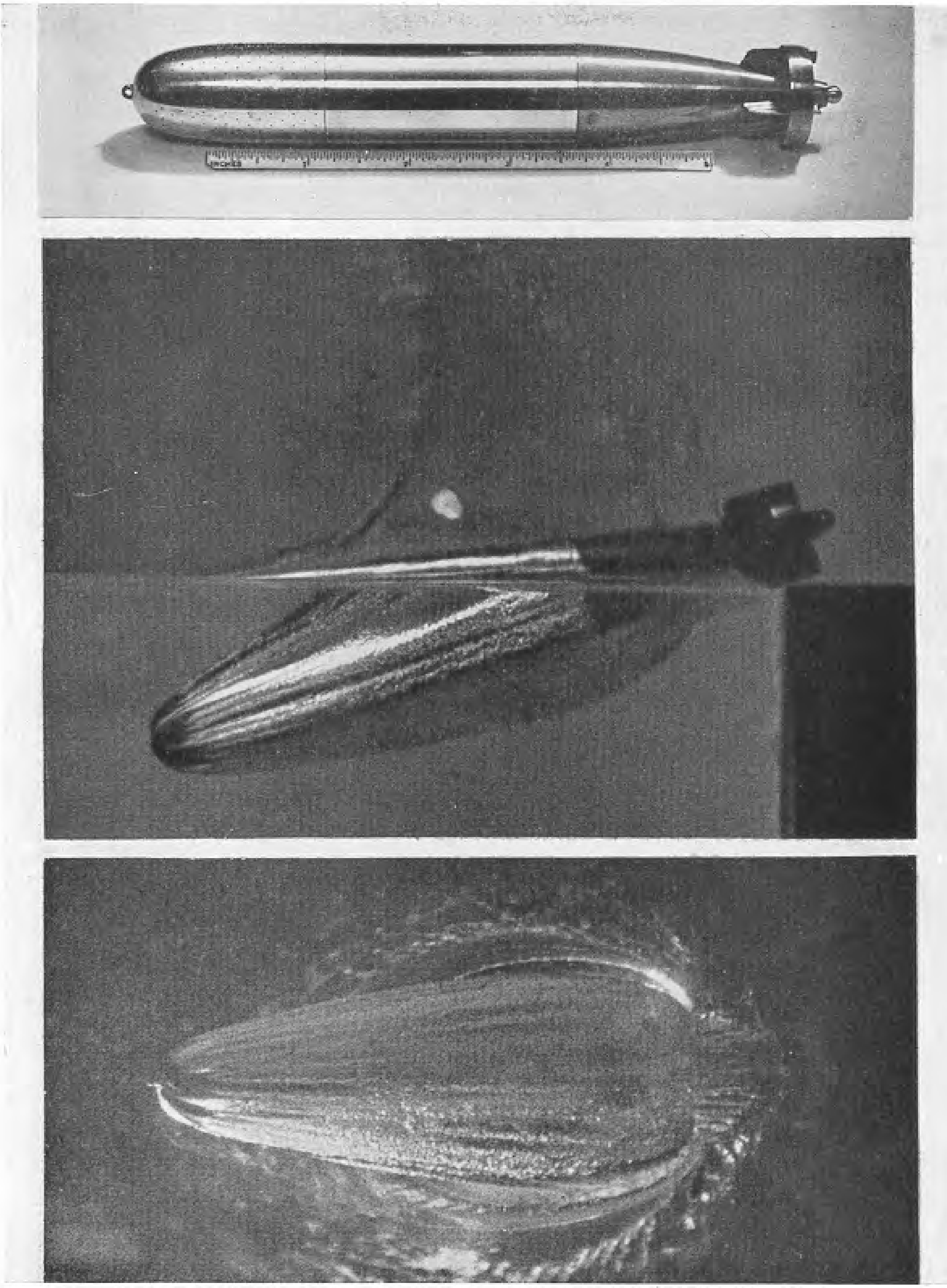

FIGURE 3. Behavior of model with vented nose. Side and bottom views of oblique entry show how venting has relieved under-pressure which produced turbulence shown in Figure 2.

6. In the laboratory at the Institute, a 24-ft by 4-ft by 4-ft glass-walled model tank providing entry velocities up to 180 fps for 1-in. or 2-in. models. This is equipped with an entry whip recorder, Edgerton-type stroboscopic lights, and a number of special types of high-speed cameras.

1.3 MODEL SCALE AND THEORETICAL STUDIES OF WATER ENTRY BALLISTICS

During the earlier part of World War II the demands on the Morris Dam facilities for study and test of service ordnance were so great that but slight attention could be paid to furthering the understanding of the hydrodynamics of the water entry of projectiles. A small glass-walled tank with a launching catapult had been set up in the laboratory and used for quantitative experiments with small models. Related studies of model behavior were conducted by the Alden Hydraulics Laboratory at Worcester, Mass., under an OEMsr-418 subcontract.

Later (in 1944) emphasis was placed on this type of work. The equipment mentioned in item 6 of the list of facilities was produced in the effort for quantitative results of precision on model behavior at water entry, including the effects of geometric scale, of entry velocity, and of other entry conditions upon the underwater behavior of projectiles. Some problems were attacked mathematically and checked by experiment. Extensive studies were made on models of the Mk 13 Mod 6 torpedo, to complement the full-scale tests carried out under Section VII of OEMsr-418. (See Chapter 2 of this volume.)

The following paragraphs outline the scope of *Water Entry and Underwater Ballistics of Projectiles*,[1] the final report [d] under OEMsr-418 on the work of Section IV.

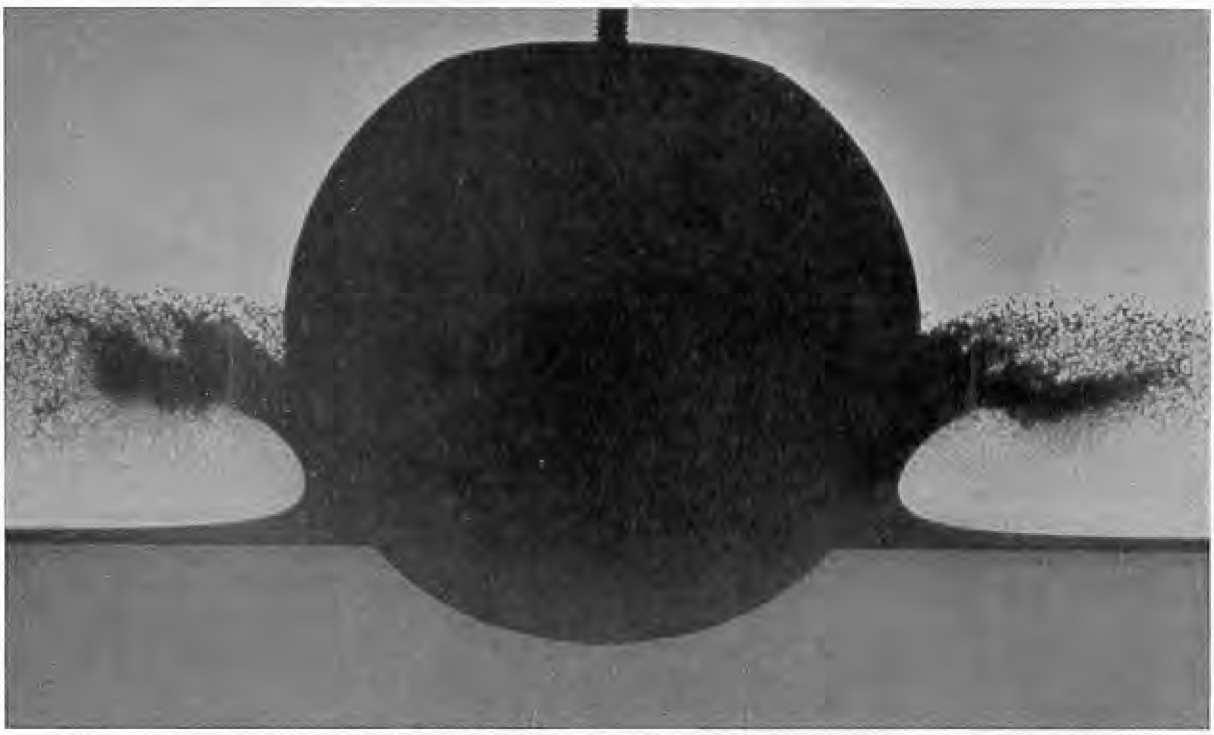

FIGURE 4. A shadowgraph, such as this one showing vertical entry of sphere, permits study of form of water surface during impact.

An understanding of the details of dynamic behavior of the entrance of projectiles into water must form the basis for the application of mathematical analysis to these complicated phenomena. These

[d] Several hundred copies of this and other OEMsr-418 final reports were distributed to the Services.

details can be studied most accurately with models of greatly reduced scale by the aid of modern high-speed photography. A matter of primary importance for practical design is the ability to predict full-scale behavior from model behavior. The report devotes considerable length to this problem. An end result is that under-pressure in an air pocket on the lower surface of the nose of the projectile at

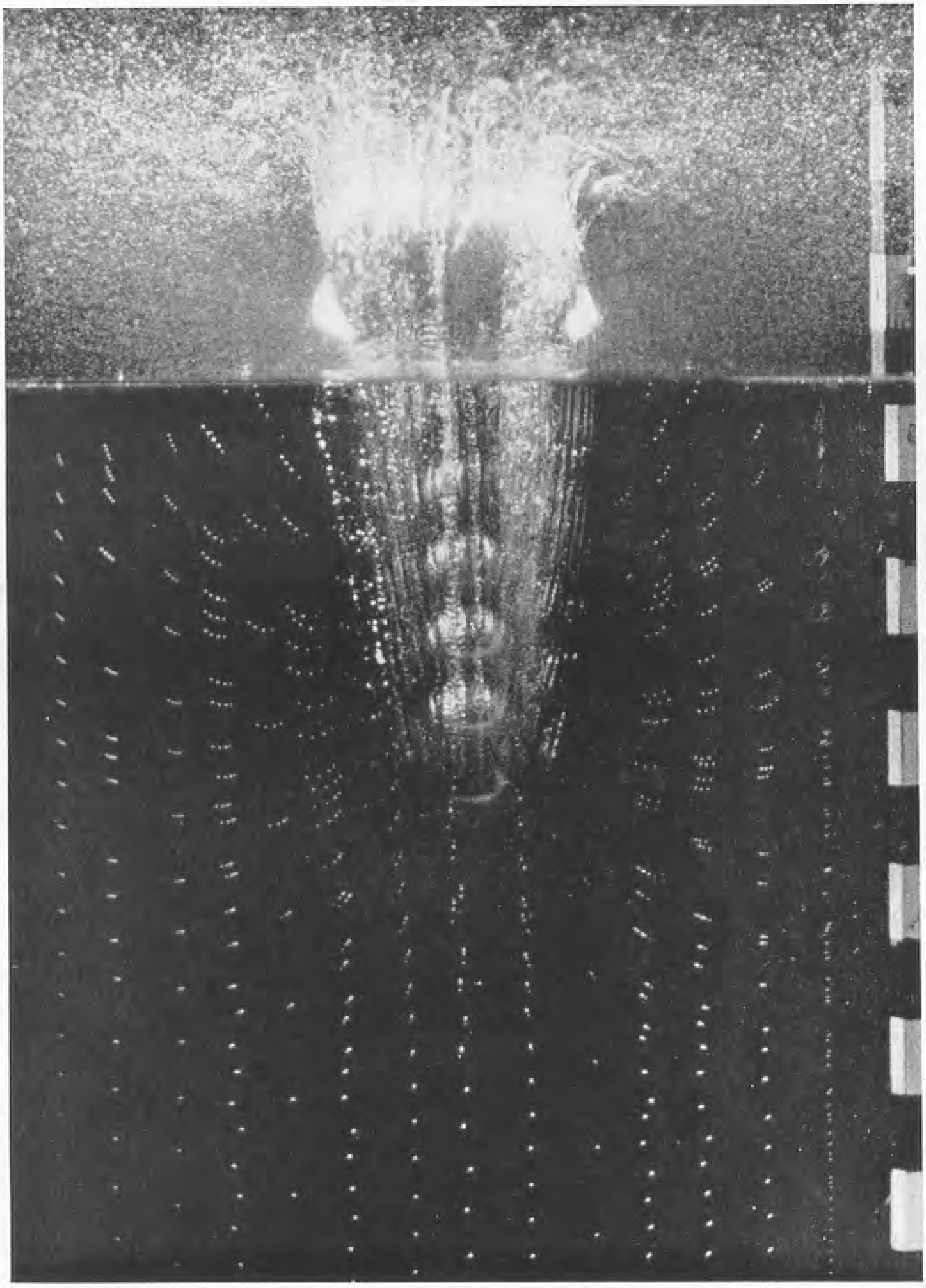

FIGURE 5. Stroboscopic methods permit study of flow patterns of fluid in neighborhood of projectiles. Here, with 300 light flashes per second, displacements of illuminated bubbles show motion of water during vertical entry of steel sphere.

[redacted]

entry plays an important role in the behavior of small models; by relieving this underpressure by "venting," satisfactory modeling may result. The effect of such venting is illustrated in Figures 2 and 3. The report also brings out the importance, in the case of high-velocity projectiles, of modeling on a velocity or "stress" basis, as contrasted with Froude scaling.

Experimental results concerning the impact deceleration of spheres and cones, the form of the water surface during impact, and the flow patterns of the fluid in the neighborhood of projectiles are reported, together with tests establishing the absence of appreciable tensional stress in the fluid during its separation from the projectile or of significant shear stress during impact. Figures 4 and 5 illustrate photographic methods that were used in making these studies.

A hydrodynamical theory of the initial stage of the vertical water impact of spheres is presented, which leads to decelerations throughout the impact stage in close accord with experiment. A dynamical theory of the underwater trajectory of a projectile in terms of a set of ten suitably chosen coefficients is included, together with a corresponding dynamical analysis of the impact phase.

Because phenomena associated with venting do not appear to have been discussed elsewhere, special attention is devoted to discussion of experimental and theoretical aspects of this important subject. Comparisons between model and prototype behavior have been presented in all cases for which adequate observational material was available. Although scale effects are apparent in the details of projectile behavior, high-velocity adequately vented models may, in general, be relied upon to reproduce the prototype trajectory within a few diameters over ten to twelve lengths of underwater travel. The behavior of Mk 13-6 aircraft torpedo models has been especially carefully studied and found to agree reasonably well with the behavior of the prototype dummy. The trajectories of 1-in. to 8-in. models of this torpedo have been obtained at entry angles in the range 12 to 35 degrees.

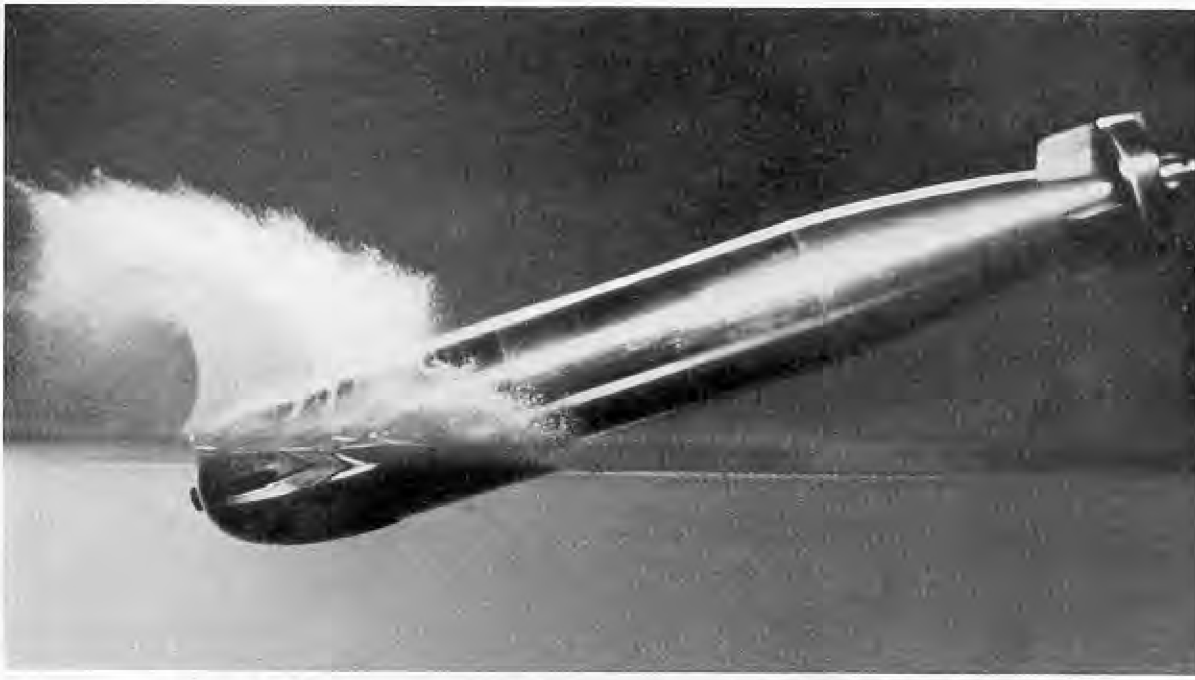

FIGURE 6. Early development of separation film of air between solid and fluid.

As an introduction to the detailed study presented in later chapters, an explanation of qualitative nature is given in Chapter 2 of the Section IV final report, by exhibiting a series of photographs of water entry which show motion of the water surface, the velocity of water particles throughout the liquid, and the form of the air cavity produced by the entry. The various stages of entry are clearly seen. First comes the impact stage, of short duration and high local pressure, during which period the water motion is set up. During this stage the water adheres to the nose of the specimen, arising shortly to form a thin splash sheath, as shown in Figure 6. Before the nose has penetrated far, usually less than one-half diameter, separation occurs between the solid and fluid, and a re-entrant cavity results, as in Figure 7. In the next stage the cavity becomes well developed about the nose and may persist with a well-defined separation point, as in the case of bluff, high-drag nose shapes, or it may tend to conform closely to the nose contours, as in the case of fine streamlined noses. In the final stages of motion, the cavity seals from the atmosphere, as in Figure 8, and gradually closes about the specimen, to be dispersed into a series of bubbles. The photographs presented in the report proceed from vertical entry of simple shapes to oblique entry of model projectiles.[e]

[e] For a much larger number of photographs of similar character, see reference 2.

Chapter 3 of the Section IV final report presents a series of trajectories of both high-drag and low-drag projectiles, and gives a discussion of the change in form to produce desired projectile behavior. Tests on antiricochet characteristics are included.

In Chapter 4 a general discussion of the problem of modeling is given.

Chapter 5 discusses nearly a dozen secondary effects which might influence model behavior. These effects were investigated briefly, for the purpose of determining their practical significance in modeling high-speed water entry. They include tank-wall effects, adhesion, tensional stress, surface condition of specimen, surface tension, externally impressed pressure and vapor pressure, compressibility of fluid, change in compressibility of the solid, gravitational acceleration and flexure of specimen. Among other things, they show the importance of the viscosity of the air in producing under-pressure under the nose of the model.

Chapter 6 presents theoretical and experimental

FIGURE 7. Subsequent development of narrow entry cavity.

[redacted]

investigations of nose under-pressure with many details on the action of the venting.

Chapter 7 presents a theoretical and mathematical treatment of the underwater trajectory of projectiles.

Chapter 8 presents a mathematical treatment of the impact of a sphere on water.

Chapter 9 gives experimental studies of the impact stage and includes the impact drag on spheres and cones, the tangential force on a sphere, the impact lift coefficient of a sphere, and the variations of entry whip with nose curvature. Observations of entry whip are by means of an optical lever system which gives high accuracy.

Chapter 10 presents a large amount of experimental results on model similitude for a variety of specimens of widely different sizes and with special emphasis upon torpedo models.

Chapter 11 consists of recommendations for a continuation of investigations of this type. It deals with experimental and theoretical procedures and the development of experimental facilities.

Appendix 1 describes the Morris Dam facilities in considerable detail.

Appendix 2 describes the model and laboratory facilities used.

The bibliography lists all Section IV reports (117) classified by subject.[f]

Throughout the book are references to the characteristics of Service types of underwater ordnance, in the perspective of the phenomena being discussed.

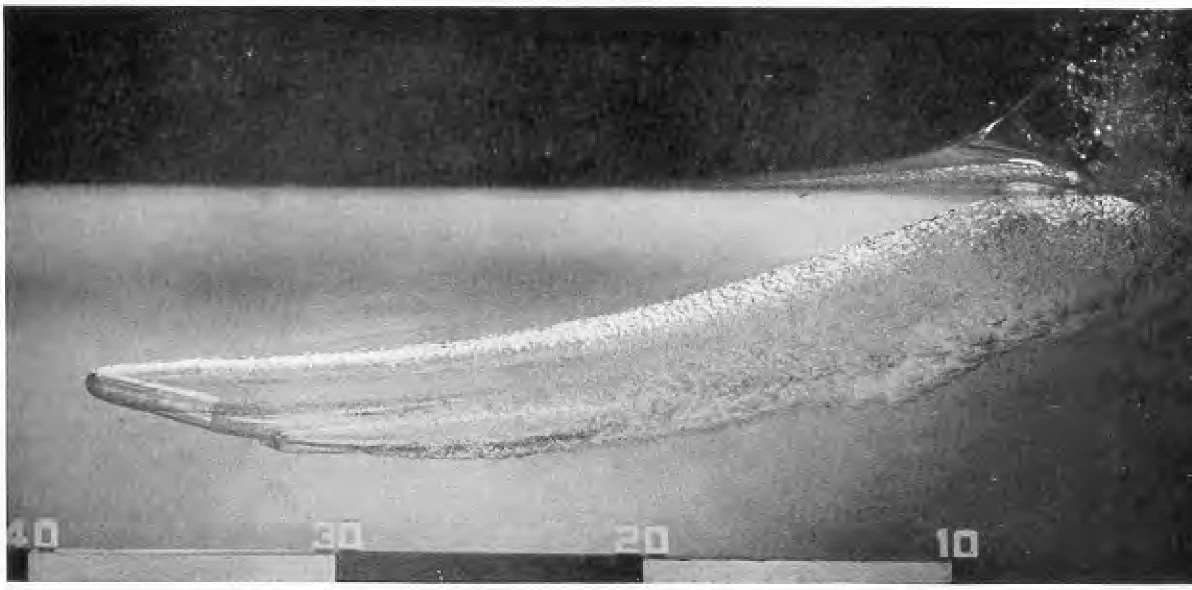

FIGURE 8. Surface closure of cavity.

[f] All these reports are included in the general bibliography appended to this volume. There they are listed by OEMsr-418 identification numbers, rather than by subject.

Chapter 2

AIRCRAFT TORPEDO DEVELOPMENT AND TESTING

By *F. C. Lindvall*[a]

2.1 INTRODUCTION

Chapters 2, 3, and 4 present in summary form the activities and results of the torpedo launching group (Section VII) which operated at the California Institute of Technology under Contract OEMsr-418. Although this Division 3 contract was concerned primarily with rocket developments, the inclusion in it of torpedo studies was advantageous. The immediate object of these studies was the measurement, in full-scale launching experiments, of the phenomena associated with the entry of a torpedo into water after release from a fast airplane at a relatively high altitude. This work, like the broader Division 6 torpedo program of which it was really a part, had as its ultimate objectives the providing of torpedo-plane pilots with more effective torpedoes and more freedom as to altitude and speed of flight at the time of release.

Out of the CIT studies came the shroud ring modification of the Mk 13 torpedo, which demonstrated in combat and in tests its superior performance under the most extreme conditions likely to be imposed by use from present types of carrier-based aircraft. Other results included substantial contributions to the design of the Mk 25 torpedo and to the general art of torpedo development. Starting from scratch in 1943, the program involved development and operation of launching facilities, of associated photographic and other equipment for recording the external phenomena of entry and underwater run, and of torpedo-borne instruments for internal measurements of stresses, accelerations, orientation, etc. Studies of torpedo control components and engineering design and structural analysis of torpedo bodies and components were also included.

Only brief descriptions of the work and its results are given in this summary. All aspects are covered completely in the final report[b] of Section VII.[1]

[a] Supervisor of Section VII (Torpedo Launching) of Contract OEMsr-418 at the California Institute of Technology.

[b] All earlier reports of Section VII are included in the bibliographies of the final report and of this Division 3 Summary Technical Report.

2.2 NEED FOR IMPROVED AIRCRAFT TORPEDOES

The U. S. Navy began World War II with an aircraft torpedo designated Mk 13. During the period between wars only limited experimental facilities were available to the Navy Torpedo Development Group, and little experience had been accumulated with this weapon. As a result, very conservative tactical limitations on altitude and speed of release had been set which were serious handicaps in combat use. Even with these limitations the water entry behavior of this torpedo and the subsequent runs were considered unsatisfactory. Early combat experience with the Mk 13 in aircraft drops was reported as discouraging. Hooking and broaching occurred with distressing frequency. There was obvious need for aircraft torpedoes which could be released at higher altitudes and higher airplane speeds with better entry and run performance. Such improved torpedoes were needed not merely for the rather slow torpedo planes then in use, but even more for effective exploitation of the potentialities of the faster aircraft then under development.

Investigations toward this end were initiated in Section C4 of the National Defense Research Committee. In the winter of 1942 to 1943, at the request of the Bureau of Ordnance, NDRC embarked on a two-pronged attack on the problem. First, the existing Mk 13 was studied with a view toward improvements in components and design by which the effectiveness of the weapon could be increased immediately. Second, a completely new design (now designated Mk 25) was undertaken. Within NDRC, general responsibility for this program was assigned to Division 6.[c] To Division 3 were assigned the essential fundamental studies of the hydromechanical phenomena associated with the entry of torpedoes into water at high speeds. All Division 3 work in this field was carried out by the California Institute of Technology, which estab-

[c] For a broader account of World War II developments in aircraft torpedoes, see Volume 21 of the Division 6 Summary Technical Report.

lished a new section (VII) for it under Contract OEMsr-418.

2.3 THE SHROUD RING TAIL

One of the most spectacular results of the CIT work was the shroud ring modification of the tail of the Mk 13 torpedo, developed in 1944.

During some of the early launchings of dummy Mk 13 units with special braces in the tail structure, it was observed that their additional drag greatly stabilized the entry, minimizing the tendency of the projectile to hook and to broach. Early work by Section IV at Morris Dam on "Mousetrap" antisubmarine rockets led to the use of a ring type of tail for stabilizing the underwater trajectory of that weapon. Model studies [d] in the CIT high-speed water tunnel under a Division 6 contract on a ring type of tail for improving the underwater stability

FIGURE 1. Shroud ring Mk 1 Mod 0 assembled on a Mk 13-2A torpedo.

of the Mk 13 torpedo led to a shroud ring having low drag in the steady run. Working with the water tunnel group, Section VII made and tested full-scale designs of such ring tails on Mk 13 torpedoes. One of these is shown in Figure 1. It is fabricated of steel with a streamlined cross section of ⅜-in. maximum thickness. The ring stiffens the guide vanes materially and affords protection to the tail structure at entry. Tests at the launching range demonstrated that these ring tails enabled Mk 13 torpedoes to enter the water at higher speeds, with less hooking and broaching, and with more stability in their underwater runs.

The California Institute converted a number of Mk 13 torpedoes to shroud ring construction for use by the Naval Air Station, San Diego, in training drops. The response of the Ordnance Officers and Torpedo Squadron personnel to the improved performance was enthusiastic. Demonstration exercises were conducted against maneuvering target ships utilizing both standard and ring tail torpedoes. In compliance with a request initiated by the Air Station, CIT made and installed a substantial number of shroud rings on torpedoes for issue to the Fleet. Subsequent tests at Pearl Harbor led to a request through the Bureau of Ordnance and NDRC for expansion of the conversion program, with the result that CIT modified approximately 1,000 Mk 13 torpedoes for service use. Many of these went immediately into combat use. Naval stations, with the aid of plans and specifications supplied by CIT, continued the conversion program on a larger scale.

Meanwhile the San Diego Naval Air Station was adding to the evidence of superior shroud ring performance at various launching speeds from 130 to 300 knots and entry angles from 20 to 35 degrees. A report circulated in July 1944, on launchings at San Diego, indicated comparative performance as follows:

	With Shroud Ring		Without	
Torpedoes dropped	218		358	
Hot and straight	204	93.5%	289	80.7%
Hot and straight, with hooks under 25 yd	199	91.2%	210	58.7%
Hot and straight, with hooks over 25 yd	5	2.3%	79	22%

The first combat action was on August 4, 1944. Continuing combat experience, some involving actions in which both standard and ring tail torpedoes were used, confirmed the superior performance of the latter, in increased percentages of hot, straight, and normal runs, when released from airplanes at higher altitudes and speeds. The ring tail modification of the Mk 13 torpedo was established as a weapon capable of withstanding any entry conditions which could reasonably be imposed by existing carrier-based aircraft.

A natural consequence of the stabilization of water entry is that the ring tail torpedo tends to dive somewhat more deeply than the original Mk 13, which is likely to make a shallow dive followed by a

[d] Summarized in the Division 6 Summary Technical Report.

severe broach. As a compensating advantage, the ring tail torpedo can be made to enter at a flatter angle without damage or serious broach and thus achieve a shallow dive. Many tests made by the Air Station at San Diego in shallow water indicated that the deeper dive of the ring tail torpedo need be no tactical handicap. Studies at the Newport Naval Torpedo Station showed that the ring could be moved forward on the guide vanes to effect a compromise between the greater stability of the aft position and the instability of the bare tail structure. The depth of dive with the ring in the forward position is somewhat reduced; however, the combat need for this torpedo was so great that the conversion program for the ring in the aft position, which was already under way, was allowed to proceed so as not to incur the delay which further testing of ring position would have required. In any event the big improvement of the Mk 13 performance came from the introduction of the ring, and changes in performance resulting from differences in ring position would necessarily be small.

2.4 OTHER IMPROVEMENTS OF THE MK 13 TORPEDO

Further studies on the Mk 13 were directed toward improvement of existing components. A good deal of study was given to structural features, heat treatment of propeller shafts, studies of bearings, and heat treatment of propeller blades. It was found that the tendency for blade bending at water entry could be reduced by proper heat treatment of the existing propellers. A good deal of study given to the problem of gyro damage resulted in a type of bearing which at the close of the OEMsr-418 work in late 1945 showed promise of withstanding 350-knot entry speeds. Attention was given to the control with the object of eliminating some of the underwater roll and malfunction of the control system in the early stages of the underwater trajectory. Much of this work closely paralleled the program of testing components for the Mk 25.

2.5 CONTRIBUTIONS TO THE DESIGN OF THE MK 25 TORPEDO[e]

The Mk 25 torpedo, which was the responsibility of the Columbia University group under Division 6, NDRC, represented a completely new design of torpedo. The launching facilities and engineering experience of the California Institute of Technology torpedo launching group were utilized to a considerable degree on the structural aspects of the problem. Various torpedo shells were tested at the launching range for damage at entry, and as weaknesses appeared design changes were made. A considerable amount of work was done on afterbody and vane construction because of the new problems created by the use of hollow guide vanes for torpedo engine exhaust. The new type of joint ring evolved for the Mk 25 also required a good deal of structural study. New propellers which were designed for this unit were also the subject of a good many launchings. Cast afterbodies of various types were tested and commercial facilities for casting experimental aluminum afterbodies were made available in the Southern California area to supplement the work which was being done in the East. As the development work proceeded, these additional torpedo components were sent to Morris Dam for launching tests, with particular attention being paid to the ruggedness of control elements of the Mk 25.

2.6 COOPERATIVE TESTS

Cooperative tests were made also for several other agencies. For the Applied Physics Laboratory of the University of Washington launching tests were run from time to time on a number of exercise heads incorporating the exploder mechanism being developed by that group. The Allegany Ballistics Laboratory requested launching tests of special propellants to discover if the shock of entry caused structural damage. Special torpedo engine igniters were tested for the Naval Air Station at San Diego and for the Columbia University group. The Westinghouse Electric Company submitted models of an electric aircraft torpedo for water entry damage studies. This work involved not only the torpedo structure itself, but also detailed studies of damage to propellers, control gear, motor, and battery. Some studies of the AAF hydrobomb were made. For the Navy, water entry tests of the Mk 1 drag ring were made with and without the streamlined nose cap which was then under study. Also, as a part of the basic research study with the Applied Mathematics Panel, launchings were made of certain special head shapes.

[e] For broader coverage of this project, see the Division 6 Summary Technical Report, especially Volume 21.

Chapter 3

BASIC RESEARCH ON TORPEDO ENTRANCE PHENOMENA[a]

By *F. C. Lindvall*

A research program directed toward more basic information associated with the phenomena of the entry of torpedoes into water was carried on concurrently with the various aspects of the development work. The water entry and behavior studies were extensive in both theoretical and experimental aspects because of the large number of parameters involved. The studies of water entry were broken into five definite stages involving various phenomena: shock stage, establishment of flow, cavity stage, transition stage, and complete immersion. The shock stage involves the water forces which are the result of an acoustic shock experienced by the body at water contact. These forces are extremely high and, because of application at an oblique angle, involve longitudinal momentum transfer as well as angular momentum transfer. These forces are of extremely short time duration, as shown both by theoretical considerations and experimental evidence. The rotating disk camera[b] gives distance-time data which are quite precise. The maximum impulsive velocity change which could occur within the limits of error of measurement with this camera are of the order of 0.5 per cent or, for typical launchings, 2 to 4 fps. From nose-mounted accelerometers and pressure plug data the magnitude of the initial shock can be determined, leading to time estimate for the duration of the acoustic shock of the order 10^{-4} second. Transverse velocity changes due to this impulsive force have also been determined to be of the order of 2½ fps. However, none of these measurements can be considered wholly satisfactory because the torpedo itself is an elastic body capable of vibration in longitudinal and transverse modes with periods comparable to the time intervals under consideration. However, the evidence is good enough to indicate that to a considerable degree the whip at entry is caused by the forces during the shock stage. Also during this shock stage, as indicated by the pressure plugs, on portions of the torpedo shell very high hydrostatic pressures exist which may cause local damage. To a considerable extent the Mk 1 drag ring tends to cushion this entry shock and minimize local damage. The shock subjects the torpedo components to high acceleration forces, but little damage results because the various components are sufficiently elastic to be self-protecting against forces of such short time duration.

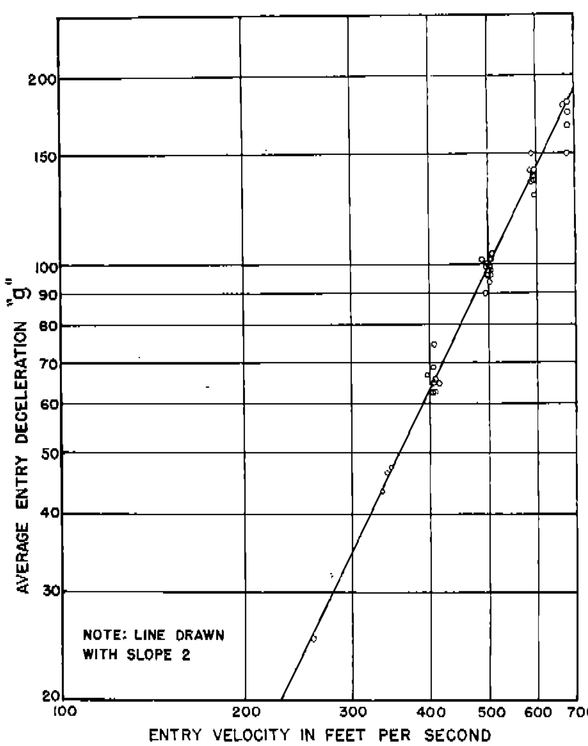

FIGURE 1. Average entry deceleration as a function of entry velocity for Mk 13 head shape (Head F).

The establishment of flow is subject to a good deal of uncertainty because of the difficulty of obtaining satisfactory detailed information during the first foot or two of torpedo travel into the water. The rotating disk camera gives deceleration information which is valid immediately after the very short time occupied by the shock stage. Typical

[a] For another discussion not limited to torpedoes, see Section 1.3 of this volume.

[b] This item, and other instrumentation, is described in Section 4.2.

data are given in Figure 1 showing a very close adherence to a square-law drag force beginning with the moment of head contact. This deceleration may be expressed as a "drag coefficient." Figure 2 shows the variation of the drag deceleration with time after entry, assuming constant drag coefficient. This drag coefficient is substantially constant for full torpedo immersion and one or two lengths of

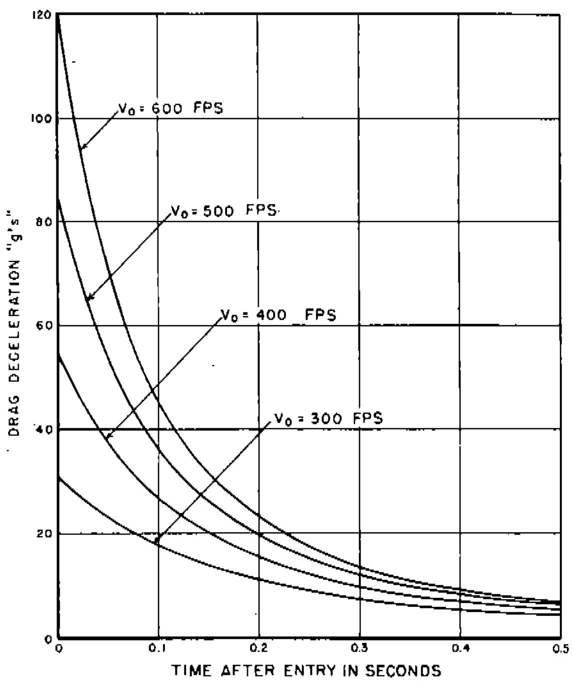

FIGURE 2. Drag deceleration vs time after entry.

additional travel. Then, because of effects occurring in the cavity stage, a higher value of drag coefficient is observed followed by the normal low body drag appropriate to fully immersed travel at speeds below the cavitation velocity. Figure 3 shows this effect.

As seen from Figure 2, the high values of acceleration may exist for 0.1 second or more. Internal components of the torpedo are thus subjected to high forces which last for periods of time which are large compared with their own natural periods. As a result, for all practical purposes, all but very flexibly mounted components are subjected to static loads corresponding to these high accelerations.

In the cavity stage of entry the torpedo is thought to be in unstable balance on its nose with the tail structure moving transversely in some direction through angular momentum acquired in the initial stage of entry. Sooner or later the tail structure encounters the more or less solid water which bounds the cavity, with resultant tail slap and application of hydrodynamic forces. The shape of the surfaces on the tail structure may cause the tail to dig into the wall of the cavity. The exact nature of this behavior is not known for full-scale torpedoes, but model studies (see Section 1.3) have indicated the performance as described to be typical. In full-scale tests the acoustic range records show evidence of this tail slap occurring well after the tail has disappeared below the surface of the water.

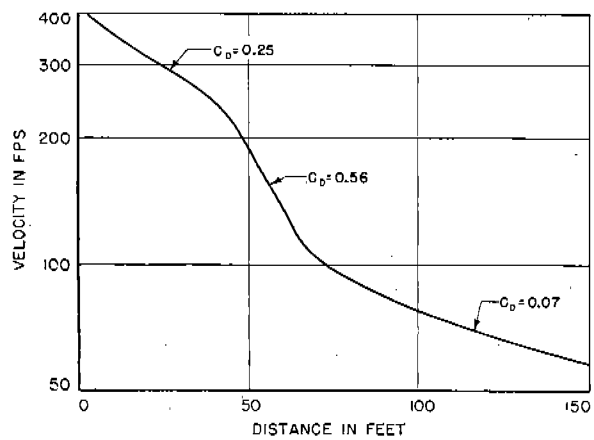

FIGURE 3. Mean velocity-distance curve for dummy aircraft torpedoes.

Pressure plug data also indicate high values of impact pressure on portions of the tail structure and the afterbody. A considerable amount of damage due to this tail slap has been observed in afterbody shells.

The transition stage from the cavity state to that of complete immersion or wetting of the torpedo can only be inferred for the full-scale torpedoes. The model work (see Section 1.3) shows the cavity to be followed by a bubble which breaks up until finally the torpedo is fully wetted. The acoustic range gives some evidence of sounds which are interpreted as bubble collapse, and general photography shows the position at which entrained air finally reaches the water surface. The observed position of rise of the bubbles correlates well with the measured information on drag coefficient change from high to low value.

In the complete immersion stage the underwater trajectories were carefully determined by acoustic range data and actual perforations of nets along trajectory. These data, together with the known

[redacted]

positions of entry and broach, if any, gave very satisfactory trajectory records of the type of Figure 4. These underwater trajectories were investigated for effect of velocity, pitch, yaw, and roll of the torpedo at entry. The data in Figure 4 show the general trend of the trajectories as affected by entry velocity. The general effect of the initial roll was slight except for conditions of large amounts of rudder setting. The effect of yaw was much the same as that of pitch, except of course in inducing horizontal deviations from a straight-line trajectory.

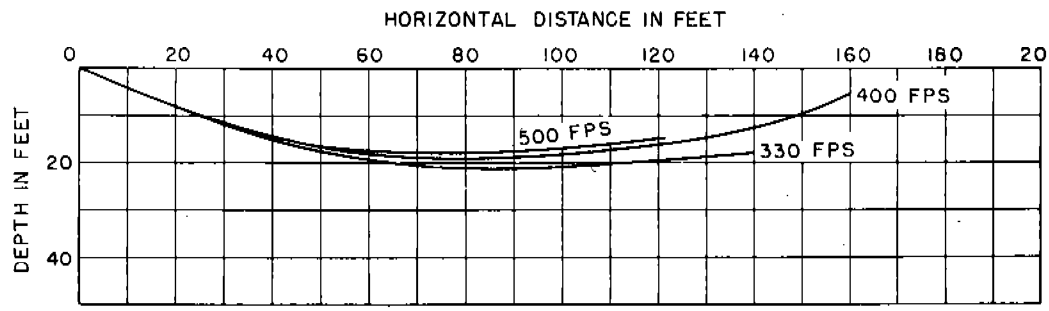

FIGURE 4. Underwater trajectories of Mk 13 dummy aircraft torpedo for initial pitch between 1 degree steep and 1 degree flat. Numbers of launchings: 330 fps, 16; 400 fps, 3; 500 fps, 7.

Pitch, that is the angle made by the longitudinal axis of the torpedo with respect to the trajectory, had a marked effect on the depth of dive or the tendency to broach. Figure 5 is typical of many sets of data taken for the purpose of showing the sensitivity of a particular head shape to the amount of pitch at entry. The data are shown in two ways: in the upper curve, the deviation of the trajectory from a straight-line projection of the airflight trajectory is measured at an arbitrary distance of 100 ft from point of entry. The lower curve gives the absolute depth of dive as a function of the pitch at entry. For the particular head shape used in these tests a steep pitch of 2 degrees or more leads to deep dives, and as much as 3 degrees of steep pitch would put the torpedo on the bottom except in very deep water. Flat pitch on the other hand leads to shallow dives, but no abnormal behavior, in the sense of an excessive broaching tendency, is indicated. Another presentation of data of this type is given in Figure 6, in which the actual trajectories are given with appropriate legends indicating the number of degrees of flat or steep pitch and the number of launchings of nearly the same amount of pitch which have been grouped as a single composite trajectory.

Although a variety of head shapes was tested, including a group of sphere-ogive combinations proposed by the California Institute of Technology Hydrodynamics Laboratory, none gave a significantly better performance than the Mk 13 head in resisting a dive to the bottom due to steep pitch at entry. Included in these head studies was one consisting of the Mk 13 shape to which was added a 90-degree cone. This cone, shown in Figure 12 of

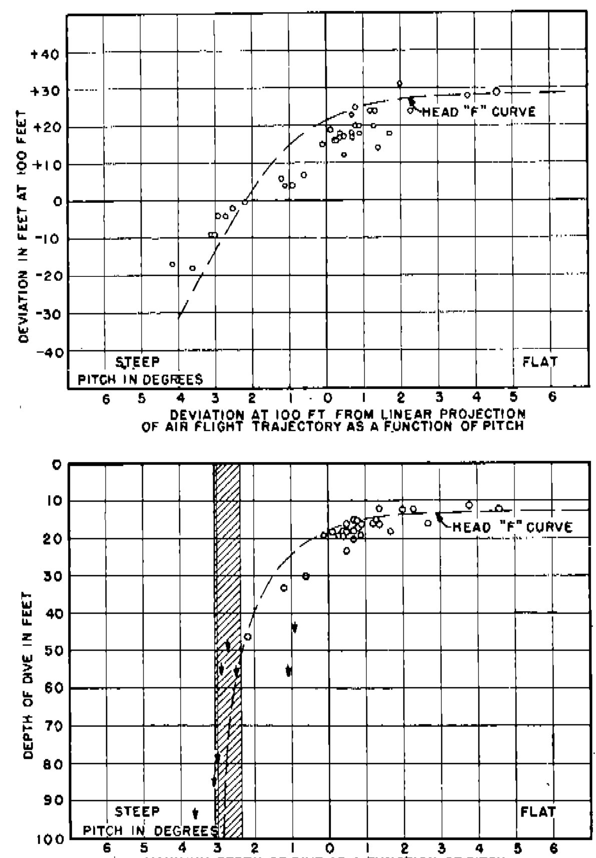

FIGURE 5. Pitch sensitivity of Head K. CIT full-scale dummy aircraft torpedo.

Chapter 4, does not improve the tendency to dive, but increases the broaching tendency for flat pitch and definitely introduces a larger whip, which is undesirable from the standpoint of structural damage. No very large departure from Mk 13 dimensions was made in any of these heads because of the overall torpedo length, which was fixed by aircraft limitations, and the necessity for maintaining approximately the same war-head volume. The variations in shape were more significant with respect to steady running drag and cavitation parameter than in modifying the pitch sensitivity.

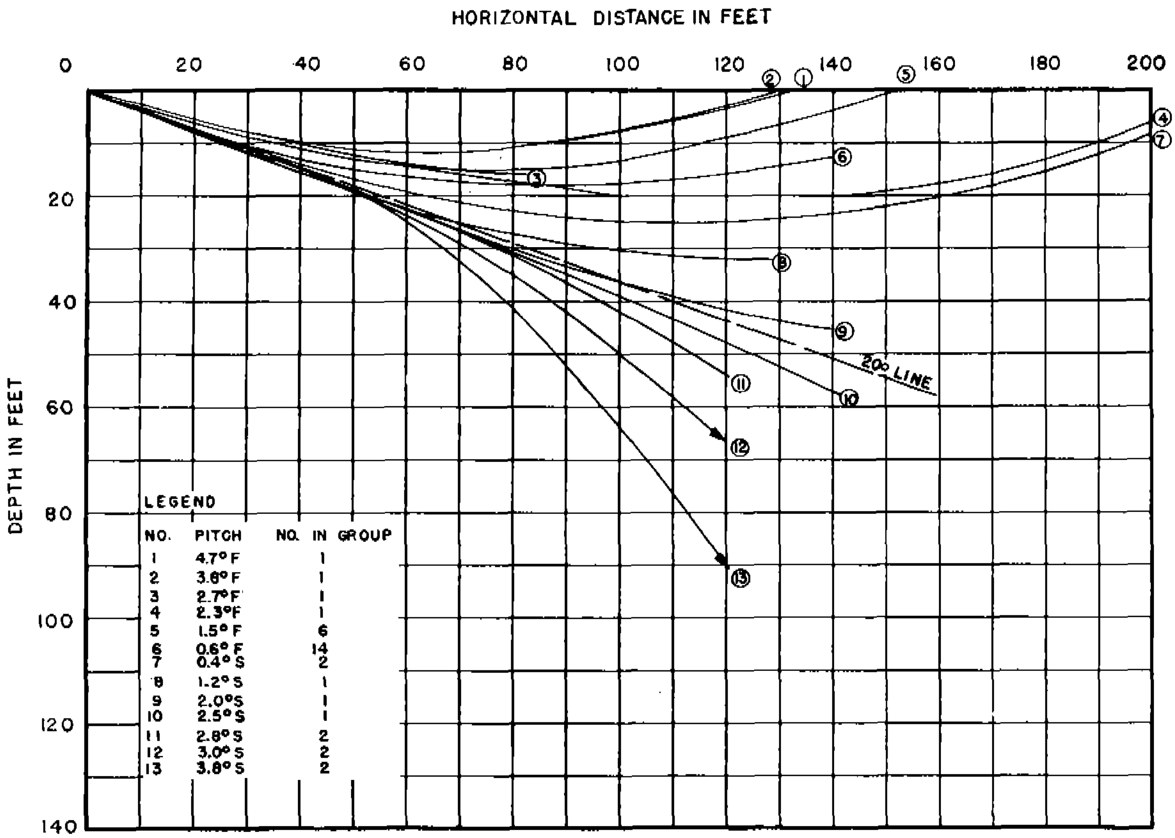

FIGURE 6. Underwater trajectory as a function of pitch for Head K. CIT full-scale dummy aircraft torpedo.

With various dummies, some of which were also used in the establishment of the underwater trajectories, the entry and underwater performances were investigated with respect to shroud ring size and reaction, rudder setting, length-to-diameter ratio, trim, and moment of inertia. The most extensive work related to the trim and moment of inertia studies. In the trim studies the center of gravity of the body was adjusted to different positions while the total weight was maintained constant and moment of inertia held fixed. The center of gravity positions were fore and aft with respect to the center of buoyancy and transversely, above and below, with respect to the longitudinal axis of the body. With the center of gravity forward of the center of buoyancy, greater entry stability was demonstrated although the underwater trajectories and depth of dive tended to be greater. In the moment of inertia studies the weight and center of gravity position were held fixed while the moment of inertia about the center of gravity was varied. The effect of the moment of inertia is not striking within the limits that are physically possible in a torpedo, but, in general, the greater the moment of inertia, the less violent are the actions of the torpedo at entry. The effect of greater length-to-diameter ratio is not entirely independent of the moment of inertia, which inevitably increases, and is similar in that trajectories are obtained which tend to follow more nearly a projection of the airflight path.

[redacted]

A large part of the study of underwater trajectories with dummies was made for the purpose of correlating, if possible, the observed performance with 1-in. models being studied by Section IV of Contract OEMsr-418. In these comparative studies the model and prototype dynamic properties were carefully scaled, and the velocities, model and prototype, were related through Froude's rule. No attempt was made in this model work to vary the pressure of the atmosphere above the water. Although it was found that the trajectories of prototype and model correlated in a general way in the early stage of the underwater run, significant deviations were observed as model velocities became low. More significant, however, was the radically different behavior of certain head shapes which, with the model, dove consistently to the bottom, while the prototype followed normal trajectories with upward curvature. Using pitch sensitivity of different heads as an index, the correlation between the model and prototype behavior was unsatisfactory.[c]

[c] Section 1.3 of this volume indicates that better correlations may be obtained by modeling on a velocity basis instead of by Froude's rule, and by venting the models.

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Chapter 4

FACILITIES AND INSTRUMENTATION FOR STUDY OF TORPEDO ENTRY

By *F. C. Lindvall*

4.1 GENERAL FACILITIES

The first problem of the CIT torpedo group was the design and installation at a suitable site of equipment capable of launching torpedoes into water at velocities and entry angles corresponding to high-speed aircraft drops. Among the requirements were sufficient water depth and length of run for adequate observation of the effects of interest. By arrangements made earlier in connection with other CIT underwater ordnance investigations, the Institute had a suitable site available only 20 miles east of Pasadena, on the artificial lake above the Morris Dam, owned by the Metropolitan Water District of Southern California. This site met these requirements in that it provided a 5,500-ft straight course of depth 100 to 140 ft. All the present and projected launching equipment is located on a peninsula approximately 3,000 ft upstream from the dam. This peninsula has a steep slope which provides a convenient support for mounting a launching tube. Steep mountains near the torpedo entry point provide excellent locations for detail and general view camera stations. The mild climate allows work to continue throughout the year, with good photographic conditions on almost all days.

Various schemes for accelerating and launching the torpedo were studied, leading to a final decision for the construction of a 300-ft tube for compressed air launching. It was believed that sufficient useful information could be obtained with a tube of fixed entry angle having the diameter of the existing torpedo to justify immediate construction of this facility, without incurring the considerable loss of time which would be required for the design of a more elaborate launcher to accommodate other projectile sizes and permit adjustable angle of entry.[a] An entry angle of approximately 19 degrees was chosen to match the general limits proposed by the Bureau of Ordnance for 350-knot airplane speed and 800-ft altitude of release. This angle was fixed with the realization that the corresponding water entry angle would probably be the lower limit of tactical operation at which satisfactory entry could be obtained and for which also torpedo damage at entry would be accentuated.

Design work on the launcher and associated facilities began early in 1943; construction of buildings and foundations at the site, early in the summer of 1943, concurrent with fabrication of launcher components. The equipment was installed during the summer and the first launchings were made in August 1943. The launching facilities have been in continuous use since that time and are now being operated on a permanent basis by the Underwater Ordnance Section of the Naval Ordnance Test Station, Inyokern. During this period the facilities underwent continuous improvement as the results of the research program dictated modifications and additions.

The general problems set for the CIT torpedo launching range were as follows:

1. General hydrodynamic effects at entry.
2. The effect of dynamic characteristics of the torpedo.
3. The effect of nose and tail structures on entry and underwater trajectories.
4. The determination of underwater trajectories.
5. The measurement of deceleration forces and the effects on structure and mechanisms of the consequent impact loadings.
6. The general structural aspects of the entry problem.

Figure 1 is a view of the range from a point directly over the launching tube. The two lines of buoys in the foreground are 100 ft apart and serve to support an array of hydrophones which constitute the acoustic range. In the distance may be seen a set of six sonobuoys which serve to extend the acoustic range for tracking the torpedo on its run. At the left in the foreground are located a control station and a camera car which is positioned

[a] A variable angle launcher of CIT design was added to the facilities after they were taken over by the Navy in 1945.

opposite the point of water entry and is moved along an inclined track parallel to the launching tube to follow changes in water level. Figure 2 is a sketch map of the facilities on the peninsula. Figure 3 is a plan and elevation of the launching tube itself in relation to the torpedo shop and working areas. thereby giving an immediate unrestricted flow of air from the tank into the tube aft of the torpedo immediately the torpedo had moved forward sufficiently to clear the inlet from the tank.

Figure 5 is a schematic drawing of the tube launching mechanism. In operation, after the breech door has been closed and ring seals (8) have been pressurized, the main gate valve (9) is opened. Any leakage into the breech section is vented through valve (5) until launching is desired. The torpedo is held in position by detent pin (7). When release is desired, handle (6) is pulled, which simultaneously releases the detent pin and seal pressure and closes vent valve (5). Air passing by the after seal (8) pressurizes the breech section, causing the torpedo to move forward in the tube and clear the Y-connection for full release of impulse air. The breech door is shown in detail in Figure 6.

FIGURE 1. General view of launching range area looking down range.

Figure 4 shows the breech end of the tube and the Y-connection to the compressed air impulse tank. A large pressure vessel of 1,100 cu ft capacity and rated working pressure of 150 psi, which was on the California Institute campus, was made available for this project to permit speedy completion of launching facilities. Later, as materials and fabrication facilities for pressure vessels became obtainable, a new tank of greater capacity and higher pressure rating was built to replace this item of Institute equipment. After a study of possible quick-acting valves for release of impulse air, a decision was made to have the torpedo act as its own plug valve,

The performance of the launching tube is shown in Figure 7, which gives calculated and measured velocities—two for the standard Mk 13 torpedo and one for a 1,500-lb dummy—at various distances along the 300-ft launching tube for three values of impulse tank pressure. Subsequently at two stations along the tube were added a series of gas booster tubes. These utilized standard rocket to the trajectory) an air scoop was added at the muzzle of the launcher which could be swung out of use or changed from top to bottom location.

The Mk 13 aircraft torpedo served as a utility instrument in many of the launching tests of this project, as well as being an object of study for possible improvement for immediate service ap-

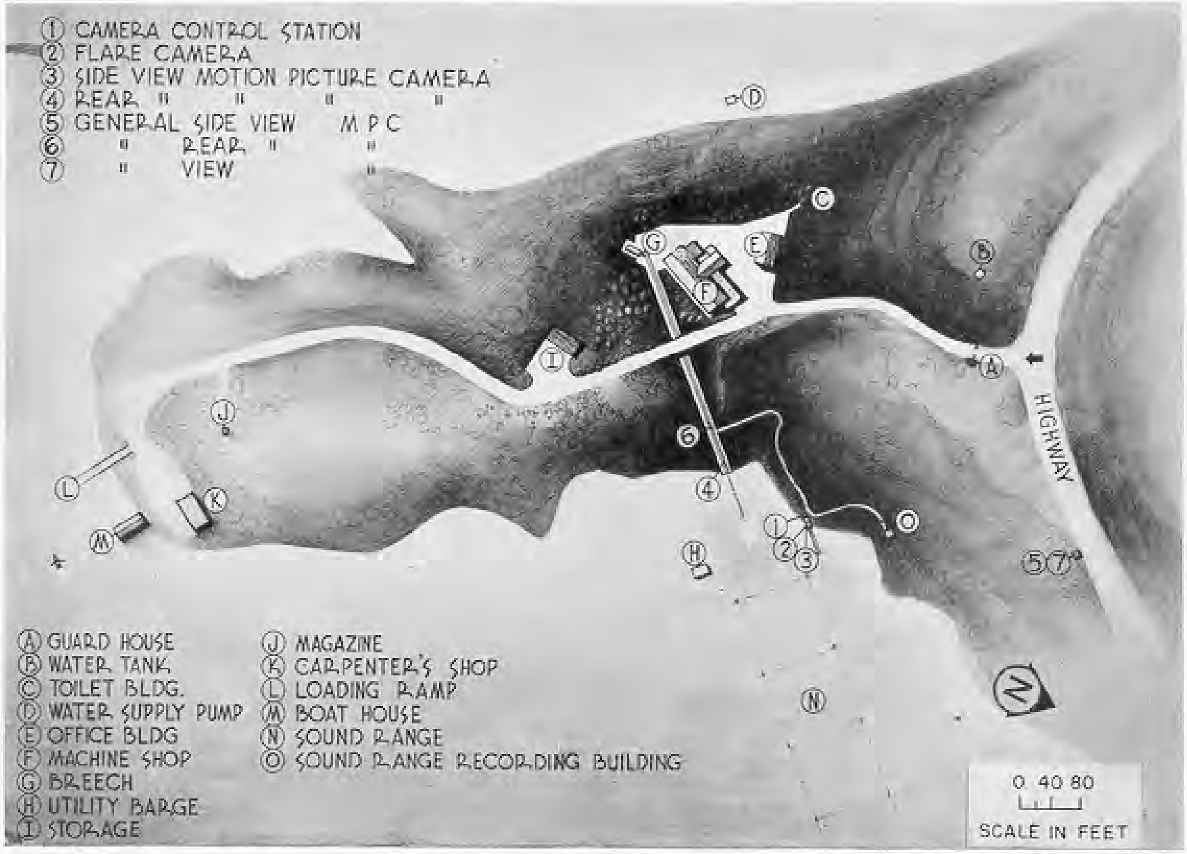

FIGURE 2. Plan of the launching site showing principal facilities. The elevation of the highway is 1,321 ft, the elevation of the working area (E, F, and G) is 1,305 ft, and the elevation of the water surface is normally 1,160 to 1,167 ft.

motors as sources of additional high-pressure gas, injected just as the torpedo passed each station. The effect of these boosters was to add some 50 fps to the muzzle velocity of the torpedo. The need for the rocket boosters disappeared with the installation of the large impulse tank, which has a volume of 1,550 cu ft and a working pressure of 350 psi.

The launching tube met the general design specifications for entry angle and velocity and put the torpedo into the water with very small amounts of random pitch and yaw. Later, to induce three or four degrees of up or down pitch at entry (relative

plication. Dummy Mk 13 units were also obtained from the Navy for some of the early launching trials, although these dummies were not usable for elaborate tests because of the impracticability of installing instruments. A number of dummy units were constructed for the project with interchangeable heads, center sections, and afterbodies so that instruments could be installed, shape modifications made, and dynamic properties of the body changed. These dummies also were made structurally rugged enough to withstand the maximum velocities of entry anticipated for this work. Dummies have been launched with entry velocities up to 800 fps.

Torpedoes are recovered in the buoyant state by boat and are towed to a landing ramp where they are floated onto a submerged trailer, which is then pulled ashore and on up to the torpedo shop. This procedure is not only rapid but is also flexible enough to follow the changes in lake elevation. A number of launchings are made with torpedoes or dummies in the buoyant condition. Other units are launched with water ballast and blowing means following general Navy torpedo exercise practice. For greater flexibility in the use of water ballast, high-density liquids are sometimes employed, the most satisfactory being a Bentonite suspension as used in the preparation of high-density mud for oilwell drilling.

Torpedoes which failed to float after launching were recovered by the 11th Naval District Mine Disposal Unit with magnetic location and diving operations. The nature of the lake bottom required a precise location before the diver was sent down.

Among the miscellaneous service facilities are a torpedo shop for overhaul and minor mechanical operations, a battery of air compressors for launching and torpedo-charging air, a small instrument and gyro laboratory, dark rooms for photographic work, a wood shop for construction of miscellaneous test equipment, a small magazine for storage of miscellaneous explosive material, and a limited amount of office space for the range supervisory personnel. In addition, a structure for housing the electronic equipment associated with the acoustic range is located near the point of torpedo entry.

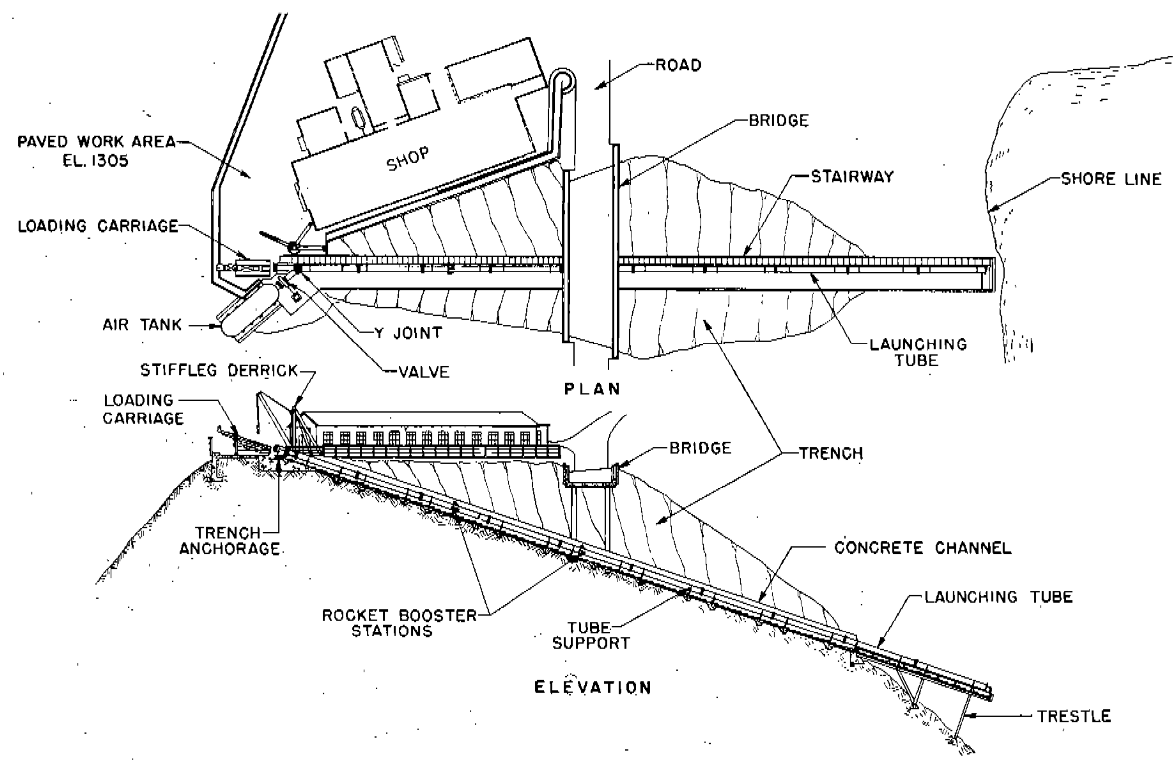

FIGURE 3. General plan and longitudinal section view of the Morris Dam Hydrodynamics Station launching equipment.

The acoustic range consists of an array of twelve hydrophones, as shown in the sketch of Figure 8. These hydrophones respond to sound impulses generated in one of the hand holes of the torpedo by detonation of electric primers set off sequentially by a timer. The responses of the twelve hydrophones to these sounds are amplified and recorded simultaneously with a twelve-channel oscillograph which superimposes timing lines on the record. From the difference in time of arrival of the sound at the different hydrophones, the position of the torpedo at the moment each sound is produced can be computed. The reduction of the acoustic data is

[redacted]

made on a mechanical computer which not only minimizes the labor of computation, but also makes the best average from the redundant data. Figure 9 gives a typical underwater trajectory in plan and elevation as determined from the acoustic range and from nets located in the range. From such a record, correlated with the sound of water impact, distance-time information is obtained from which velocity and underwater deceleration may be derived.

4.2 INSTRUMENTATION

4.2.1 External Observations

Measurement of the deceleration of the torpedo as it enters the water is of fundamental importance. Internal recording equipment for deceleration measurements involves many difficulties of application and of final interpretation of results; consequently, except for local effects within the torpedo due to impact or shock loading, an external measurement of the behavior of the body as a whole is most satisfactory. During the entry phase external photography can follow the entry for a distance representing roughly 80 per cent of the body length. Ordinary motion picture camera technique gives inadequate time resolution and ultra-rapid cameras are inapplicable because of insufficient light. A new technique therefore was developed for this work, consisting of the use of a small but intense light source mounted on the rear structure of the torpedo, the image of which was recorded on a photographic plate every thousandth of a second by the chopping action of an interrupting disk in the light path of the camera. This disk had narrow radial slits carefully

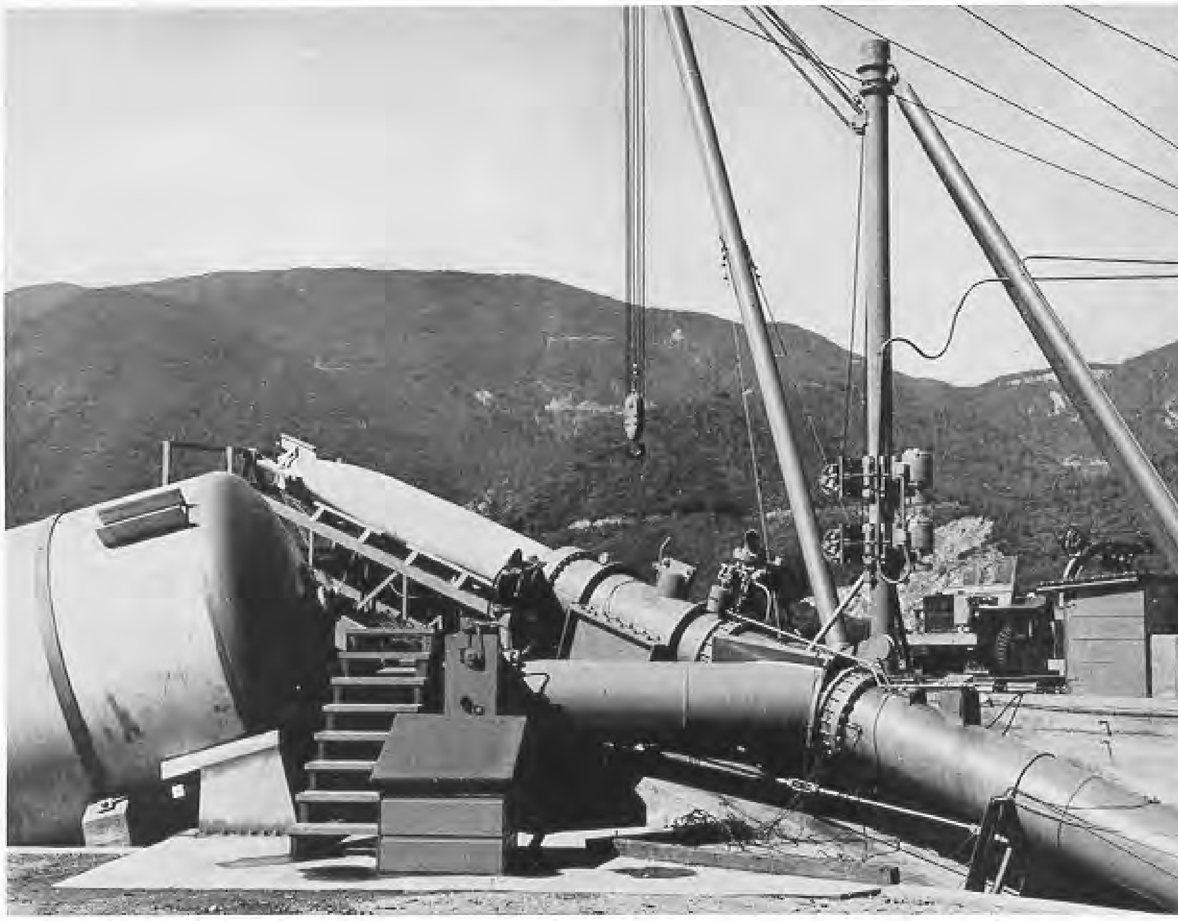

FIGURE 4. General view of breech end of launching equipment. This illustration shows the No. 1 impulse tank and the breech and Y sections prior to reinforcing.

spaced around the circumference which permitted the camera to see the light source on the torpedo a thousand times each second for intervals of approximately 20 microseconds each. A typical record obtained with this camera is shown in Figure 10, in which two light sources were employed on the tail of the torpedo. The general illumination of the background is sufficient to bring out reference marks which aid in the reduction of data, but the total time during which the camera shutter is open must be kept to a minimum to avoid overexposure of the background. In so far as the essential record of torpedo position as a function of time is concerned, the images of the light source on the torpedo are sufficient and could be obtained at night just as well. The light sources used consisted of small steel cups with suitable mounting brackets. These cups were slotted and packed with an aluminum powder pyrotechnic mixture which was ignited at the time of torpedo release by means of an electric primer and a bit of black powder paste. This arrangement gave a brilliantly illuminated slit approximately ⅛ in. wide and 1½ in. high. The original 5-by-7 glass photographic plates, from one of which Figure 10 was reproduced, were measured with great precision on a measuring engine such as is used with spectrograms. The camera was located approximately 70 ft from the point of entry, and the precision of measurement was such that the position of a good flare image could be determined at the torpedo to within a tenth of an inch.

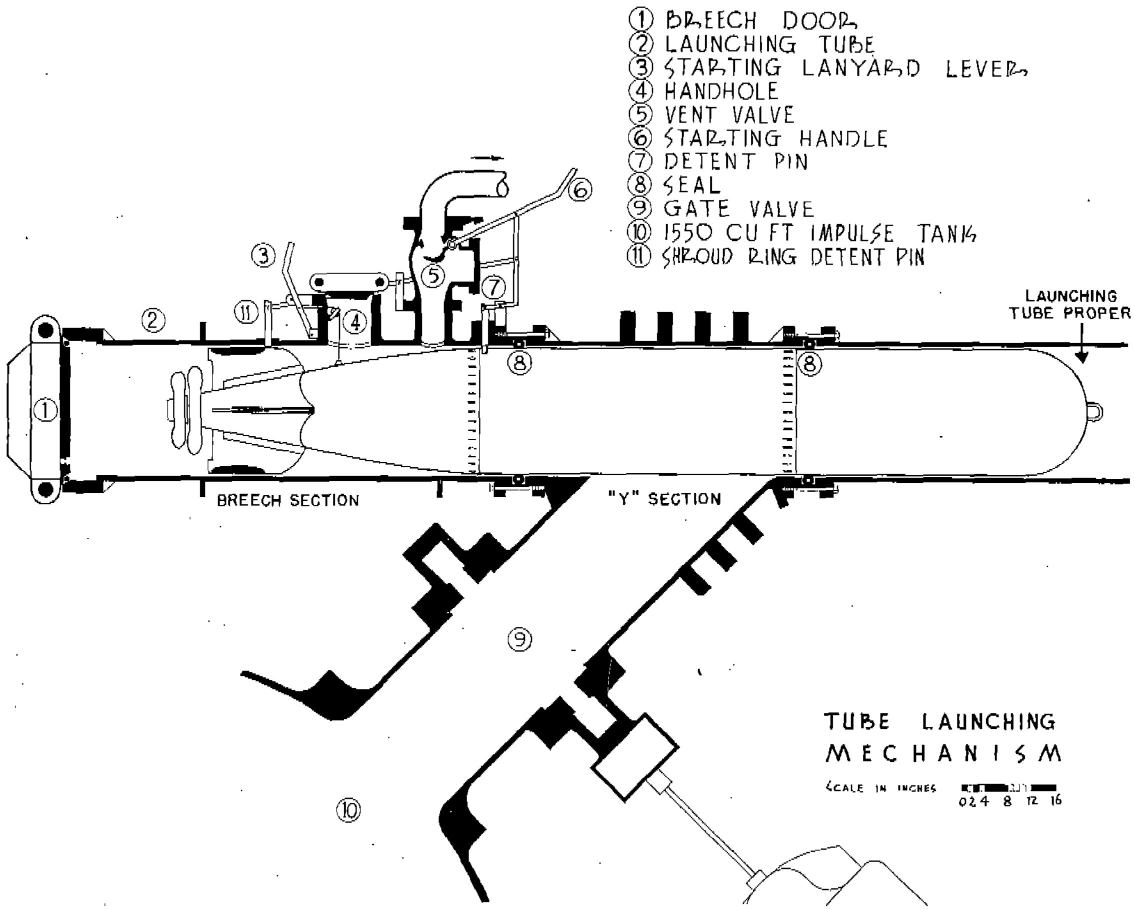

FIGURE 5. Sectional view of breech end of launching equipment. Structures connected with openings 4 and 5 are actually located 90 degrees toward reader from position shown.

By measuring the intervals between flare images and plotting these measurements against time, a velocity-time curve (Figure 11, upper curve) is obtained for the entry of the torpedo up to the time the tail disappears from view. These velocity curves form a straight line parallel or nearly parallel to the time axis until the torpedo strikes the water; at this

[redacted]

point the line joining the points bends abruptly downward. The slope of this portion of the velocity curve is proportional to the deceleration of the torpedo. When two flares are used on the torpedo, one above the other, a line joining the upper and lower flare images is a measure of the angle of the axis of the torpedo. Thus both the pitch angle and change in pitch angle of the torpedo can be determined.

Other photographs of water entry were made with motion picture cameras located in various positions. A Mitchell 35-mm high-speed (125 frames per second) motion picture camera was located adjacent to the rotating disk camera and gave in considerable detail from this side view the aspect of the torpedo prior to entry and the behavior during entry, as shown in Figure 12. The synchronously rotating timing disk in the foreground gives a time scale which permits a photographic check of the velocity of entry to be correlated with the muzzle velocity as measured with an electronic timer.

Directly under the breech of the launching tube a 16-mm, 64-frame-per-second camera is located to record the rear view of the entry phenomena. From these records may be determined the roll and yaw.

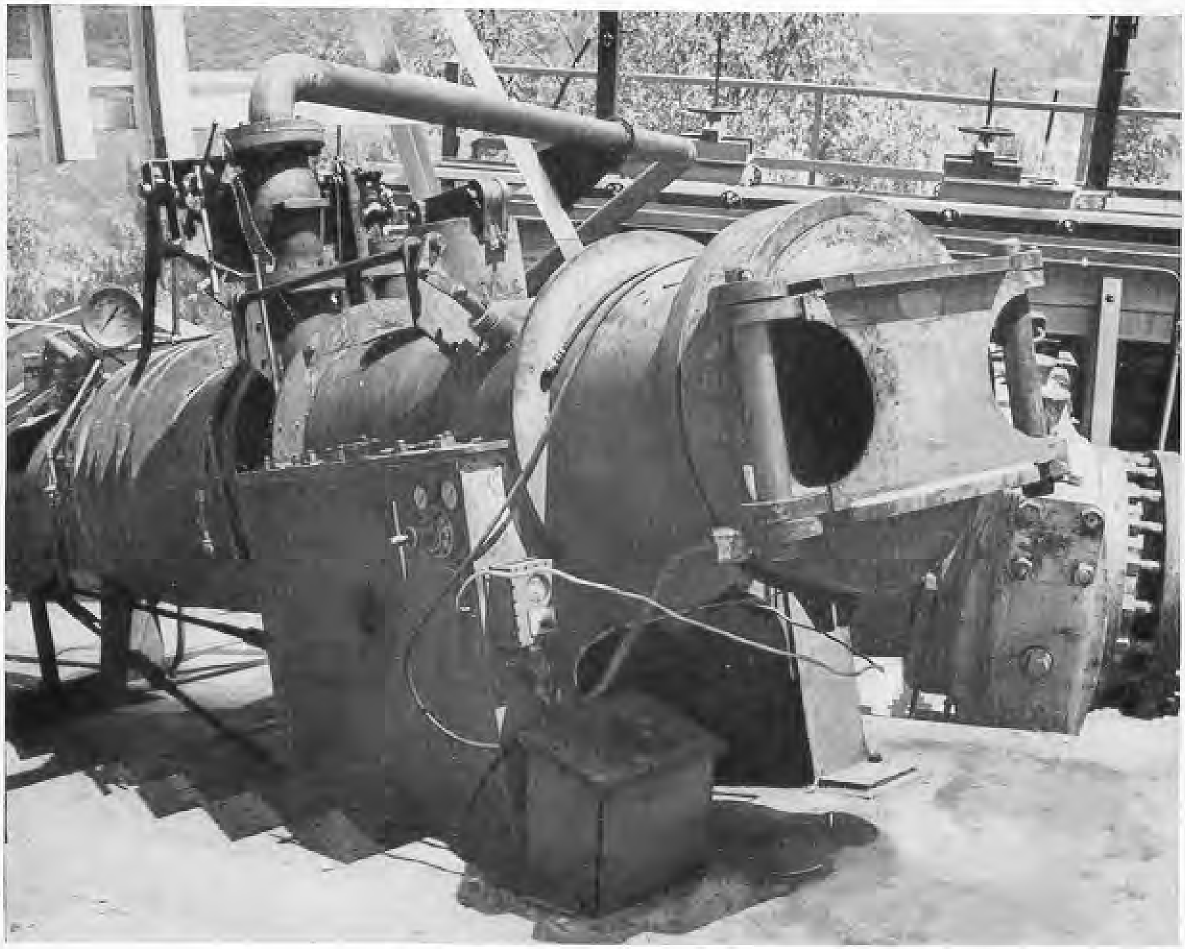

FIGURE 6. View of breech end of launching tube.

A general view motion picture camera is situated on the hillside approximately 400 ft from the centerline of the range. The field of view is about 400 ft at the center of the range. This camera is used to record general data such as positions of entry, broach, and re-entry, the velocity at broach, length of underwater run and of air travel, the angle at broach and re-entry, and of hook at broach, the height of the broach, and the path of run following re-entry. A timer in the field of view of this camera has four disks driven by a synchronous motor at speeds of 1,500, 150, 15, and 1.5 rpm. Film from this camera is viewed for measurement with a single-frame projector and a system of plane mirrors which puts the image on a measuring grid ruled in perspective to represent the true coordinates of the lake surface and aligned by placing the images of the range buoys in coincidence with their respective positions on the grid.

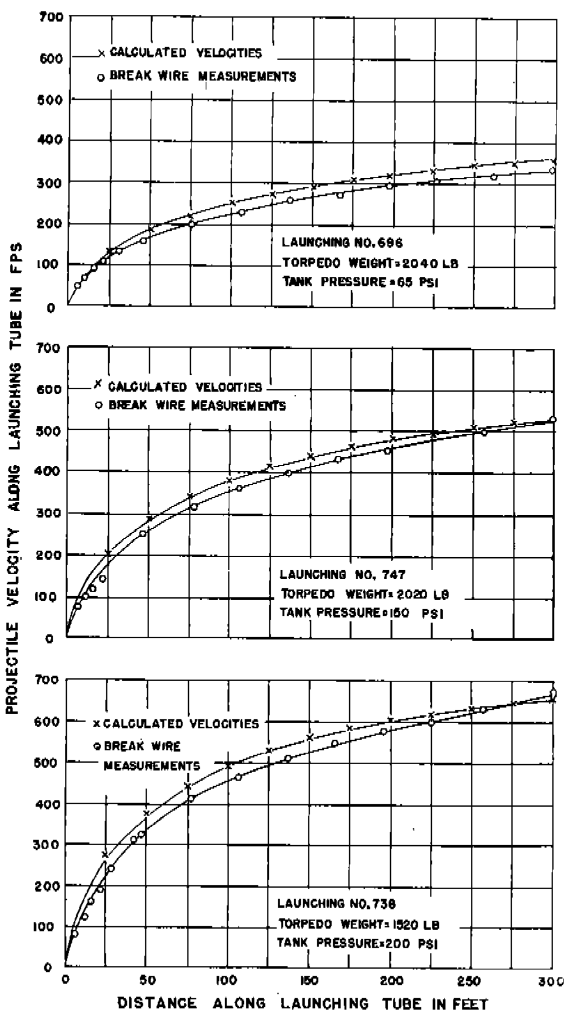

FIGURE 7. Calculated and measured projectile velocities along the launching tube.

An overhead camera may be used, if desired, in a camera car on a cable suspension system directly over the range, permitting location of the camera directly over the point of entry. The camera car and camera mechanism are remotely operated from the central camera control station.

Underwater photography has been used in an experimental way with motion picture cameras installed in watertight submerged drums. Only at certain times of the year is the clarity of the water sufficient to permit photography of full-scale torpedoes, because of the distance the camera must be located from the line of the underwater trajectory in order to keep a field of view great enough for more than a single torpedo length.

A general rear view camera is used to record powered runs of the torpedo. This is a single-exposure camera which, from a height above the lake surface, photographs the track of the torpedo at any desired stage in the run. The photographs are measured by placing them over a transparent grid so that the coordinates of the torpedo track may be determined.

4.2.2 Internal Measurements

Internal instrumentation included a variety of devices for obtaining acceleration of torpedo components, pitch, roll, propeller speed, control positions, time of water entry, and miscellaneous events to be correlated with the moment of water entry. The heart of the recording system for these various instruments was a specially designed neon tube camera, as shown in Figure 13, using a $1/25$-watt neon bulb as the essential element. Three models of this recording camera have been constructed and used. Each unit consists essentially of a bank of neon bulbs, an optical system which projects the light onto moving motion picture film, a film drive, a vacuum tube oscillator which periodically flashes one of the neon tubes and thus produces a timing reference trace on the film, and switches which start and stop the camera. In the various models batteries are either self-contained in the camera or are placed in an auxiliary box in the torpedo. All this equipment must be extremely rugged in order to avoid distortion or damage resulting from the severe shock of high-speed water entry. The neon bulbs are either on or off depending on contact position in the instrument whose operation is being recorded. A typical record obtained with this camera attached to a step accelerometer is shown in Figure 14.

Accelerometers

This step type of accelerometer consists of a series of cantilever springs, shown schematically in Figure

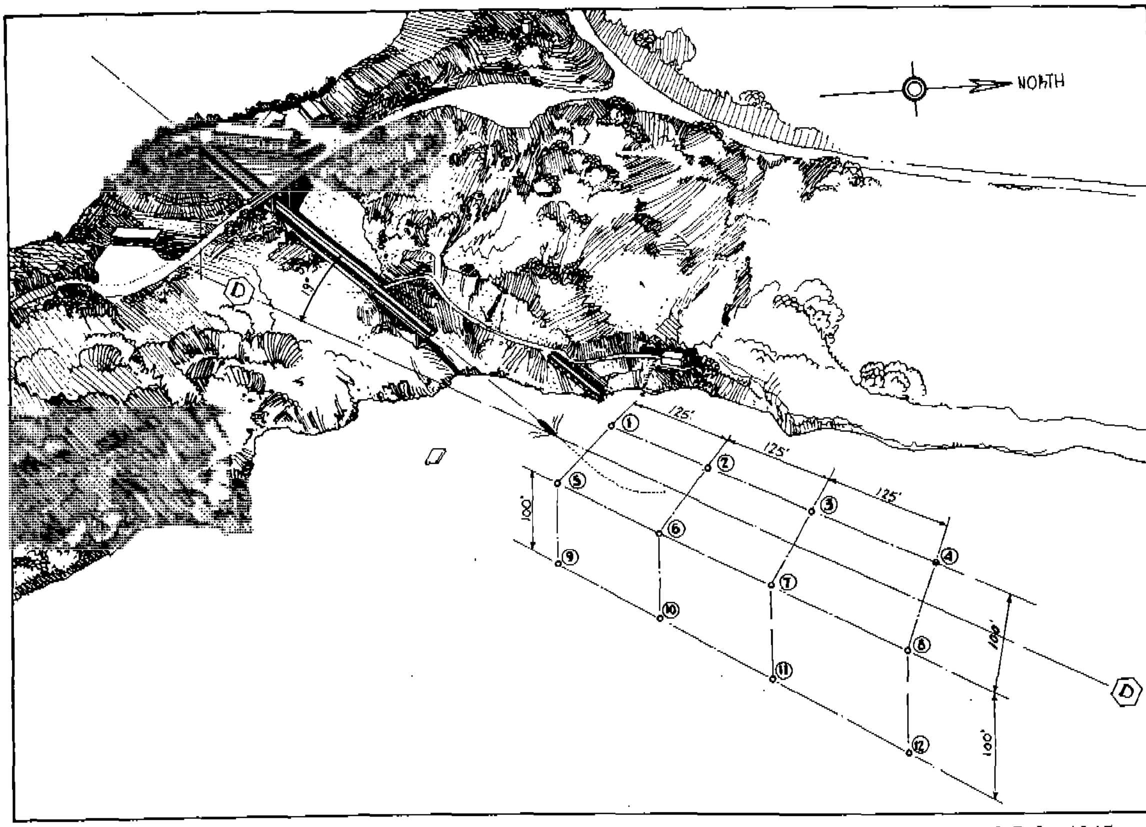

FIGURE 8. Aerial perspective showing torpedo launching area and sound range as of July 1945.

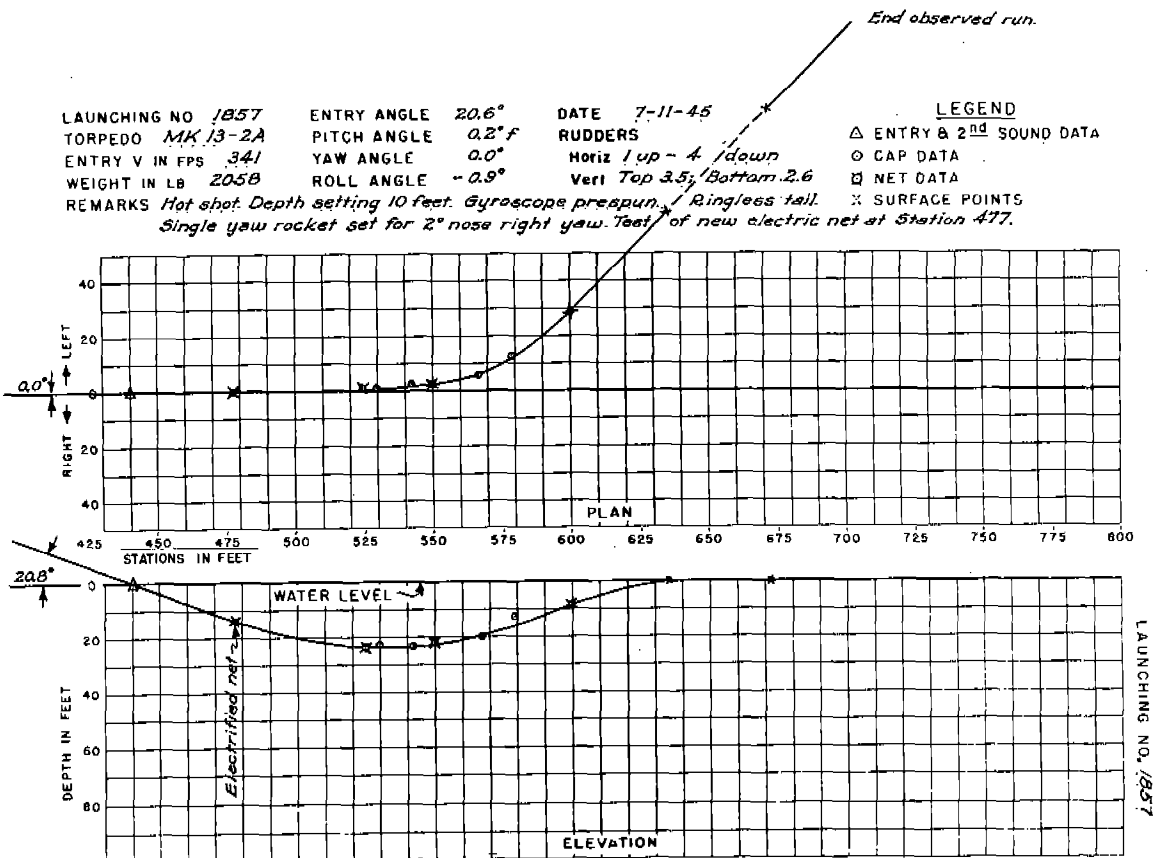

FIGURE 9. Typical trajectory plot.

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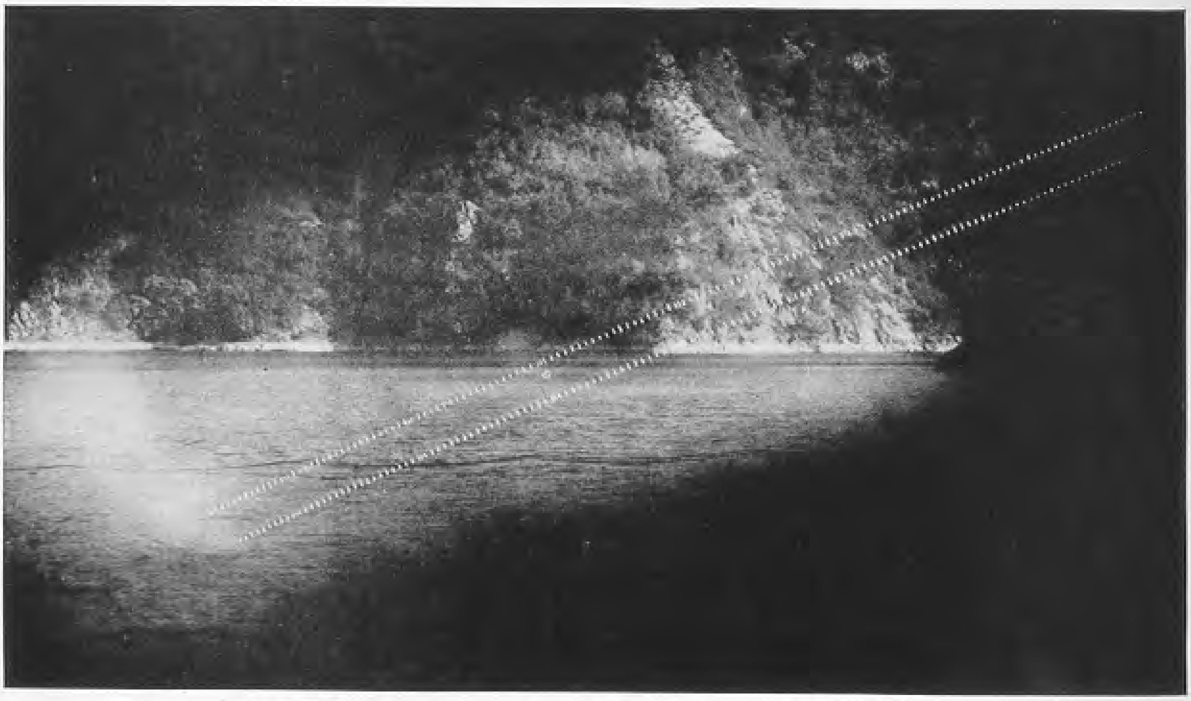


FIGURE 10. Flare camera record with two flares on torpedo tail.

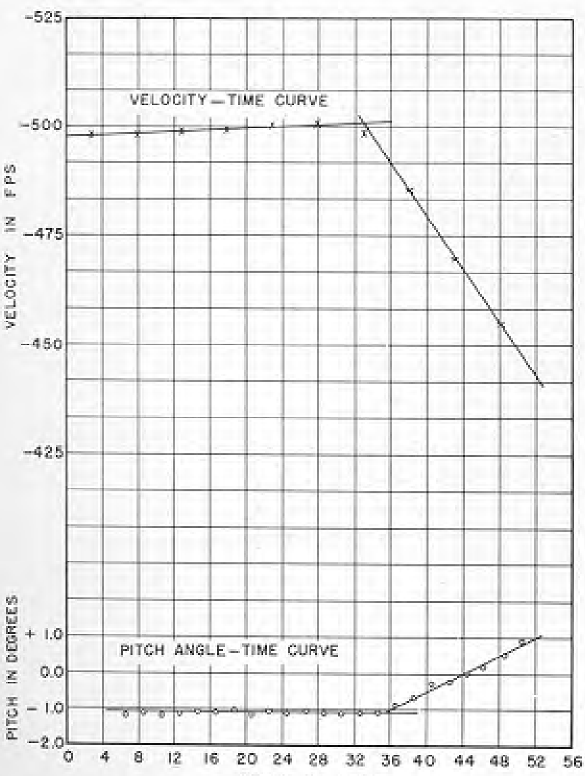


FIGURE 11. Typical velocity-time (upper) and pitch-time (lower) curves derived from flare measurements.

15, preloaded to break contact under a specified value of acceleration and each controlling one of the neon camera bulbs. The record of Figure 14 is actually for three separate accelerometers located in

FIGURE 12. Entry of Mk 13 torpedo with added 90-degree nose cone.

the torpedo at the three positions shown in the sketch. The record shows the 500-c timing marks, auxiliary 100-c timing marks, and the responses of the 21 neon tubes connected to the three accelerom-

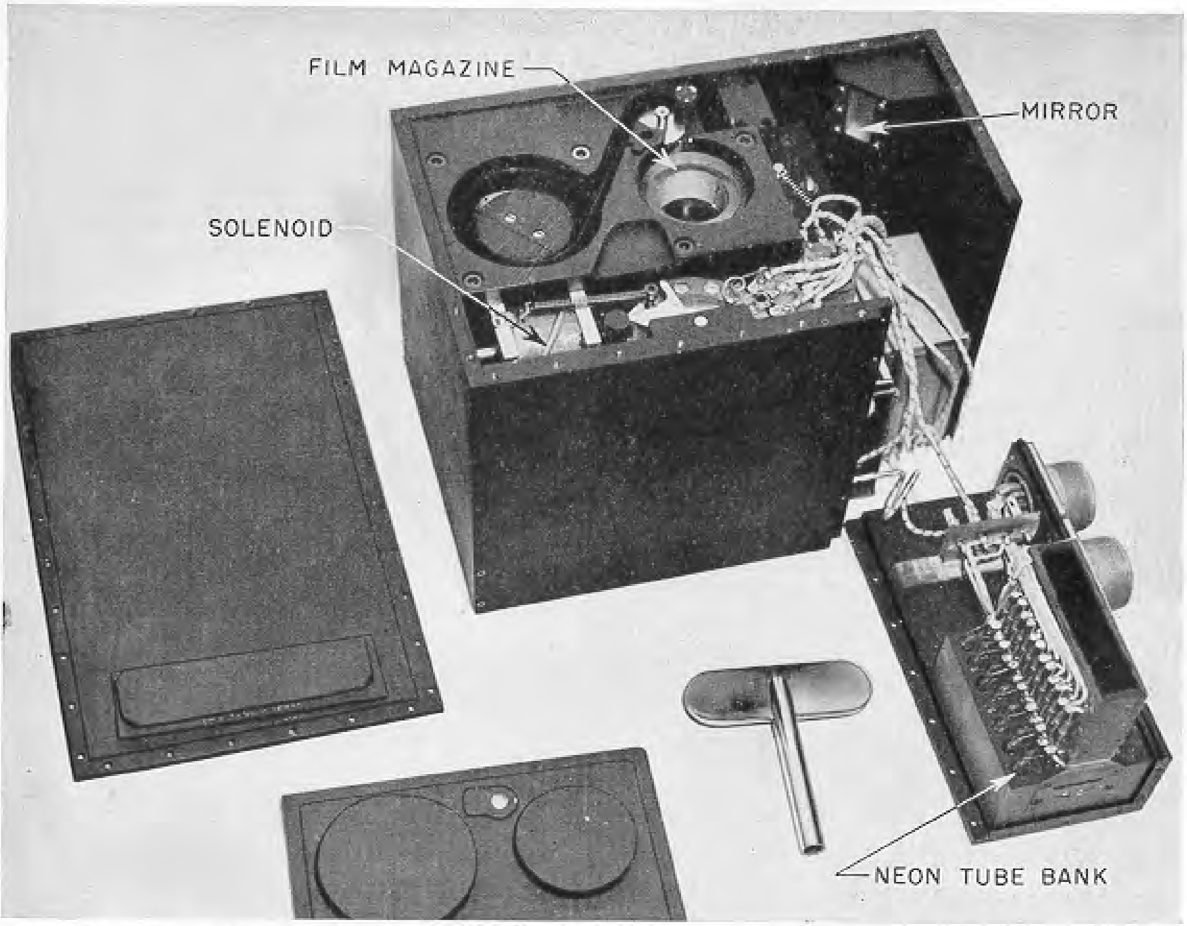


FIGURE 13. Neon tube camera—Model 3. Top and bottom plates removed, showing removal of neon tube bank.

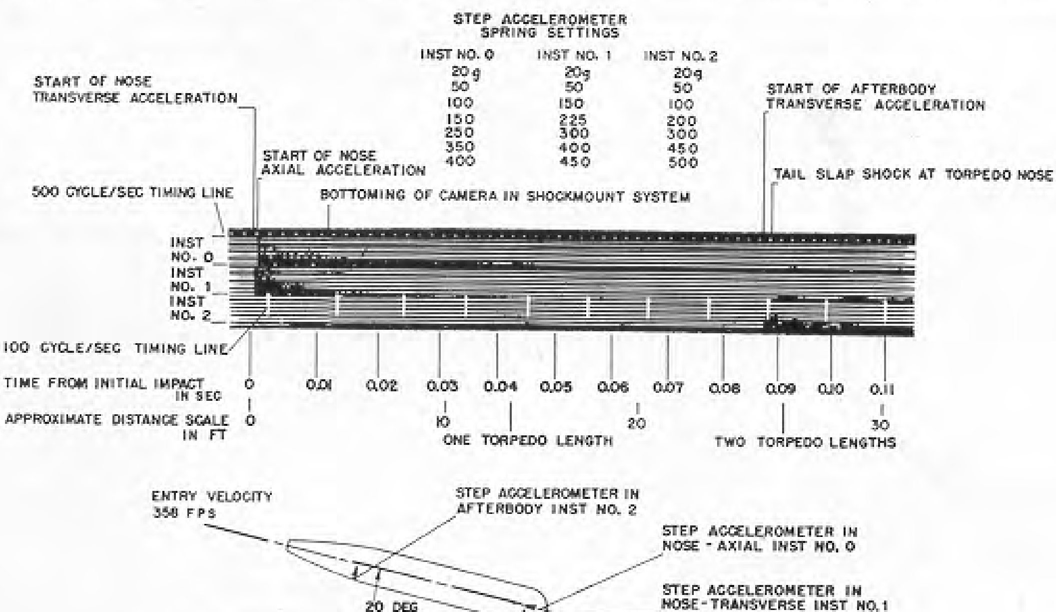


FIGURE 14. Typical step accelerometer record from a launching test.

C[redacted]

eters. From this record can be obtained the time duration of the various magnitudes of acceleration shown by the blacked-out portions of the neon tube records. The approximate distance scale shown on the figure is derived from external photographic and underwater acoustic data.

Various types of accelerometers of the indenter type and the copper ball deformation type have been used in the course of this work. However, the recording step accelerometer is by far the most satisfactory, because it not only gives a time record, but also records a sequence of repeated shocks as contrasted with a single record resulting from a deformation or displacement type of instrument which gives no time history and which may be quite ambiguous as a result of repeated shocks of unknown character. For certain types of testing, the simpler instruments of the indenter or deformation type may be calibrated against the step accelerometer and then used with some confidence for subsequent accelerations of the type for which the calibration is valid. The step accelerometer lends itself to analysis and reliable calibration so that peak values of acceleration may be measured with confidence and repeated shocks determined reliably, provided the repetition rate is slow compared with the natural frequency of the spring elements in the accelerometer.

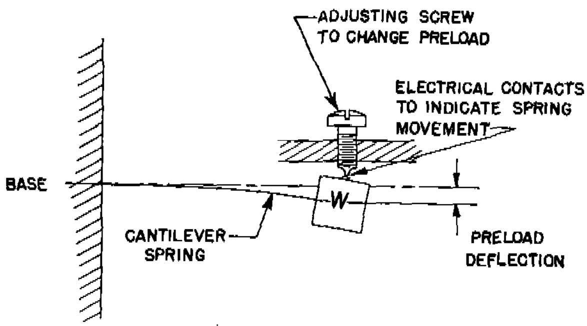


Figure 15. Representative CIT accelerometer spring element.

Accelerometer Calibration

To provide a calibrating system for accelerometers, a drop table with control and recording equipment was constructed. A table carrying a standard accelerometer, to which other apparatus or accelerometers for calibration purposes could be attached, was arranged on guide rails to have a free fall of approximately 20 ft onto buffers, dash pots, lead plugs, or other suitable stopping means. The standard accelerometer consisted of a spring-mass system in which the spring was a thin-wall Dural cylinder to which was attached wire strain gauges determining the deflection of the spring system. The unit was calibrated statically and dynamically. The dynamic calibration was made by the sudden release of a known weight suspended from the bottom of the accelerometer. This procedure caused the same resistance change in the strain gauges as sudden loading, though with opposite sign. The natural frequency of this accelerometer was approximately 5,000 c and was valid therefore for acceleration measurements on phenomena of frequencies up to 1,500 c at least. The electrical output of the strain gauges was amplified and recorded on a moving film oscillograph consisting essentially of a film drive and a cathode ray oscillograph beam swept in only one direction. A record of a step accelerometer calibration made with this equipment is shown in Figure 16.

Damage Instruments

Additional dynamic studies were made in the torpedo models with what were called "damage instruments." These instruments consisted of simple mechanical structures, cantilever beams, and tension specimens, which were loaded by acceleration forces. Figure 17 illustrates the tension type. These were for standard tension specimens whose properties were known from static tests on similar components. They are secured at one end to mounting structure and loaded by acceleration forces acting on the weight attached at the other end. The weights are loosely guided in the enclosing cylinders. Figure 18 indicates an array of cantilever members, some of which are loaded with definite weights applied at the ends, others of which are uniformly loaded by the acceleration forces. As a result of a particular launching some of these test members are undamaged, others have taken permanent set, and for some of the tension specimens actual failure may have occurred. The information resulting from these tests is helpful in designing structural components of the torpedo to withstand the shocks of water entry.

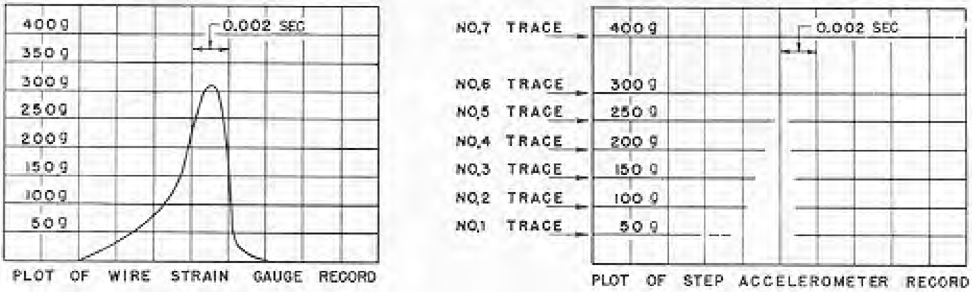


FIGURE 16. Comparative plots of the wire strain-gauge accelerometer and step accelerometer records of the same acceleration pulse.

Additional damage information was obtained through the use of scratch gauges of the de Forest type and from bonded-wire strain gauges. The use of the wire strain gauges was limited to tests in which external recording could be used, whereas the de Forest strain gauges were installed at many points in the torpedo and at various points of its mechanism to determine the maximum strains resulting from launchings.

FIGURE 17. Tension-type damage instrument.

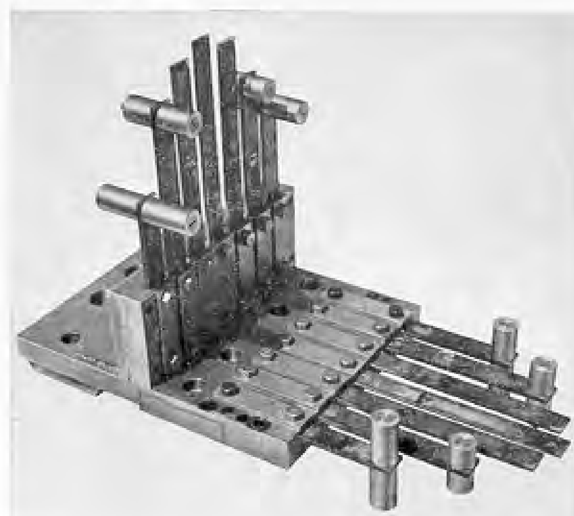


FIGURE 18. Model No. 1 bend-type damage instrument.

Hydro Pressure Plugs for Localized Peak Pressures

The typical accelerations for the torpedo as a whole determined from these various instruments lead to an interpretation of water drag forces proportional to the square of the velocity, but certain local damage effects are traceable to transient water pressure forces of much greater magnitude. To study this effect hydropressure plugs were developed, as shown in Figure 19. These plugs, of ½-in. overall diameter, were inserted at various stations in the torpedo shell as indicated in the sketch of Figure 20. The recording is done by permanent set of an annealed phosphor-bronze diaphragm. This permanent set is correlated with hydraulic pressure studies of identical disks. As seen in the tabulated data of Figure 20, extremely high hydrostatic pressures occur during entry which correspond quite well to the calculated pressures which are the product of the local torpedo velocity at impact and the acoustic impedance of the water. At various points on the torpedo body,

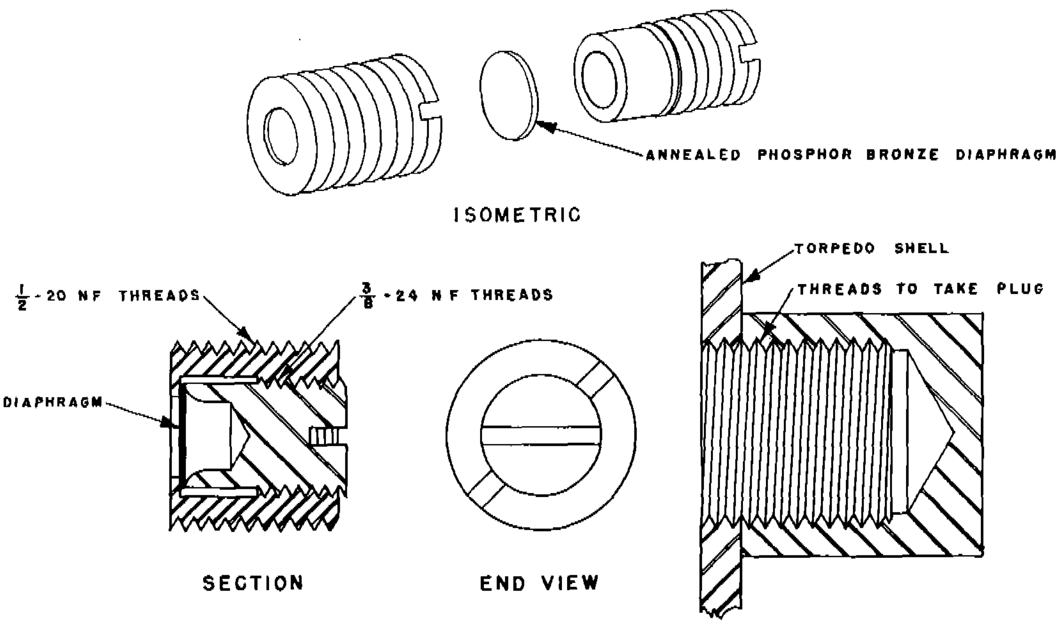
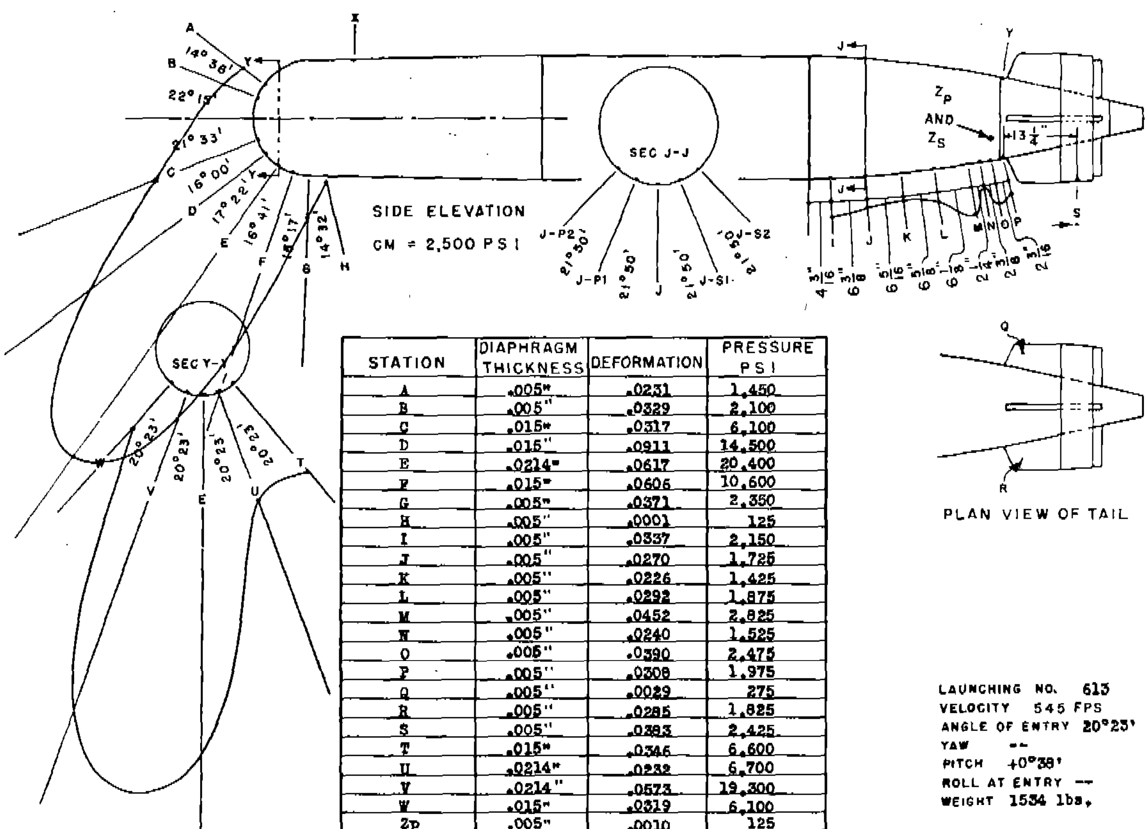


FIGURE 19. Hydropressure plug and suggested mounting method.



STATION	DIAPHRAGM THICKNESS	DEFORMATION	PRESSURE PSI
A	.005"	.0231	1,450
B	.005"	.0329	2,100
C	.015"	.0317	6,100
D	.015"	.0911	14,500
E	.0214"	.0617	20,400
F	.015"	.0606	10,600
G	.005"	.0371	2,350
H	.005"	.0001	125
I	.005"	.0337	2,150
J	.005"	.0270	1,725
K	.005"	.0226	1,425
L	.005"	.0292	1,875
M	.005"	.0452	2,825
N	.005"	.0240	1,525
O	.005"	.0390	2,475
P	.005"	.0308	1,975
Q	.005"	.0029	275
R	.005"	.0285	1,825
S	.005"	.0383	2,425
T	.015"	.0346	6,600
U	.0214"	.0232	6,700
V	.0214"	.0573	19,300
W	.015"	.0319	6,100
Z_P	.005"	.0010	125
Z_S	.005"	.0000	125 OR LESS

FIGURE 20. Typical data sheet and pressure distribution plots obtained with hydrophone units in a launching test of a Mk 13 dummy torpedo at 20-degree entry angle and 550-fps entry velocity.

[redacted]

particularly in the area of nose contact with the water, the geometrical volume displacement of the water requires a water velocity in excess of that of the acoustic velocity. Consequently, the water is compressed, and high local pressures result. Similar effects, due to slap on the side of the entry cavity, occur in the afterbody and in portions of the tail structure. While these effects are of very short time duration, they frequently give rise to local shell damage. Figure 21 indicates the effect of the Mk 1 drag ring (pickle barrel) in reducing these localized high pressures. Although the drag ring, a light wooden structure used in standard service drops, was intended originally for stabilizing the airflight of the torpedo, experience at Newport has shown beneficial results in the reduction of damage to torpedoes.

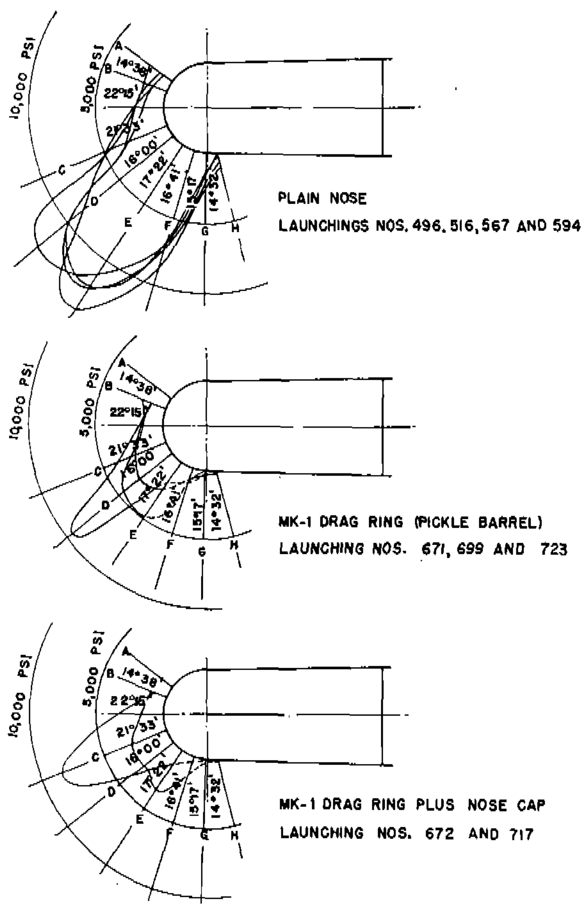


FIGURE 21. Hydropressure plug data showing peak pressure distribution on torpedo nose with various coverings. Entry angle 20 degrees. Entry velocity 410 fps.

Orientation Recorders

A gyroscopic orientation recorder was developed to determine the orientation of the torpedo with respect to its trajectory. Knowledge of this orientation is of great importance in determining the subsequent motion, beginning with the precontact stage and continuing through the steady running phase. This instrument was designed to give roll, the angle of rotation of the torpedo about its longitudinal axis; pitch, the angle between the trajectory and the torpedo axis in a vertical plane; yaw, the similar horizontal angle; attitude, the angle between the longitudinal axis of the torpedo and any horizontal plane; and deviation, the angle formed by the intersection of a vertical plane through the longitudinal axis of the torpedo and the vertical plane through a set course. The instrument consists of a Mk 12–1 gyro so modified that rotation between the outer gimbal ring and the torpedo and between inner and outer gimbal rings may be recorded. The instrument is contained in a cubical case, and the gyro is so oriented that the spin axis and the outer gimbal axis will always be released at 90 degrees to each other. The gyro is held by a centering pin which can be disengaged by the starting acceleration of a tube launching or by a solenoid.

The gyroscopic orientation recorder measures angle changes which are recorded on the neon tube camera in the torpedo by means of very light brushes sliding over commutators. The angular resolution of a camera commutator is determined by the spacing of the contacts and is 1 degree between the outer gimbal and the torpedo, whereas between the inner and outer gimbals it is 0.75 degree. The stepwise record of contact closure obtained from the camera film is reduced to data of the type given on the roll record of Figure 22. Roll records of this type are much more satisfactory than those obtained with the Foxboro depth and roll recorder, because the gyro is not subject to inertia forces. The pendulum of the Foxboro instrument responds to centrifugal force which exists during any hooking or turning of the torpedo and gives a spurious indication of roll. Furthermore, the gyro instrument is not limited to the 30-degree travel of the Foxboro pendulum.

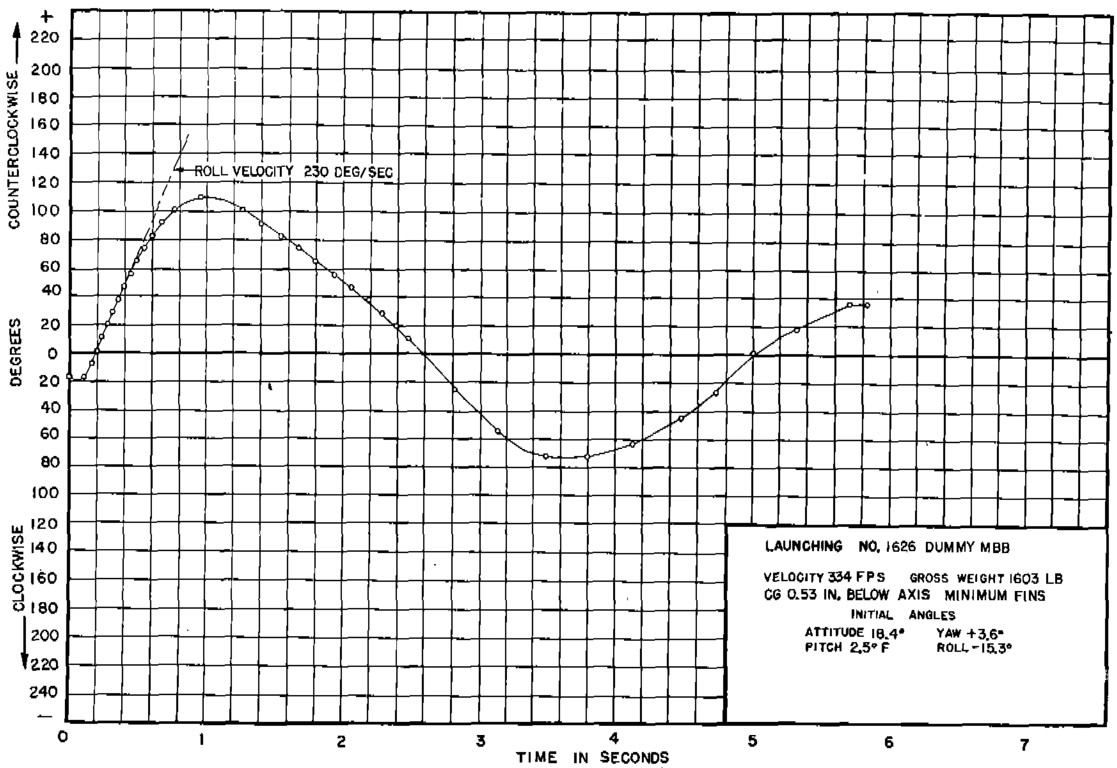


FIGURE 22. Roll-time record. No propellers.

Many other accessory instruments and measuring techniques were utilized in the project, but for such detail reference should be made to the general report on this project.

PART II

ROCKET PROPELLANTS AND INTERIOR BALLISTICS

By *B. H. Sage* [a]

During World War II, artillery rockets were again employed to advantage in a number of special tactical situations. This renewed interest in rockets may be ascribed in part to the greater mobility of arms and the consequent premium placed upon a low ratio of weight of launching equipment to weight of ammunition fired per unit time. The development of rockets for the U. S. Army and Navy was initiated in 1940 by the National Defense Research Committee. The discussion in Part II summarizes the status of the interior ballistics of artillery rockets, their ignition, and the utilization of dry-processed double-base powders as propellants. There are also a few brief statements on the overall situation regarding the propulsion systems of such rockets and the probable future course of progress in this field.

The material presented in Part II arises almost entirely from the development and experimental production activities of Section V of Contract OEMsr-418 between the Office of Scientific Research and Development and the California Institute of Technology. This work was carried out between October 1941 and October 1945, for the most part by the professional members of Section V. Primary emphasis was upon the designing, construction, testing, and semiproduction fabrication of relatively simple propulsion systems of artillery rockets. Little, if any, effort was made to achieve rockets of the highest performance, since the necessary meticulous refinement in design would have materially increased the time required for their development and decreased the number of rounds which could have been prepared with the limited facilities available.

The marked emphasis which was placed upon the development and experimental production of specific weapons prevented the systematic collection of as large a background of experimental facts concerning the underlying principles of interior ballistics, ignition, and deflagration of double-base propellants as would normally be expected in the course of a program of comparable scope carried out under less urgent conditions. Nevertheless, sufficient information has gradually been accumulated to permit a number of significant generalizations to be made, which are presented in some detail in two book-length publications.[1,2]

No further explicit reference will be made to these books, which in themselves represent a summary of the subjects under discussion and which serve to a large extent as the basis for the present limited treatment. On the other hand, specific references will be made whenever possible to the technical reports of Section V which contain the pertinent experimental data. Also all reports issued on the work of the section have been listed.[b]

The technical progress which was realized by Section V represents the efforts of at least 20 professional men. However, the work of W. N. Lacey and D. S. Clark of the staff of the California Institute of Technology was particularly helpful in connection with the supervision of certain of the activities. Acknowledgment should also be made of the significant contributions of R. N. Wimpress, W. H. Corcoran, and Q. Elliott to the field of interior ballistics and propellants, and of the assistance rendered by B. H. Levedahl and D. F. Botkin in the studies of physical and thermal properties, respectively.

[a] Supervisor of Section V (Propellants and Interior Ballistics), Contract OEMsr-418, California Institute of Technology.

[b] In the general bibliography appended to this volume the principal Section V reports are listed under OEMsr-418 designations IAC, IBC, ICC, IDC, IGC, JAC, JBC, JCC, JDC, and JGC.

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Chapter 5

INTERIOR BALLISTICS

By *B. H. Sage*

5.1 PRINCIPLES OF ROCKET PROPULSION[a]

THE PRINCIPLES OF ROCKETRY, which are relatively simple, have been known for an extended period of time. In a general way, the relationship between the exterior ballistic behavior of the round and the performance of the rocket motor can be indicated in the following manner for a projectile containing a weight W of inert components and a weight w of propellant traveling at a velocity V with a thrust F applied. Under these circumstances, using t for time and g for acceleration of gravity, the acceleration is given by

$$\frac{dV}{dt} = \frac{Fg}{W + w}. \tag{1}$$

The weight of propellant changes as burning progresses and the products of combustion are expelled through the nozzle. If air drag and other minor effects are neglected, the velocity at the end of burning, V_0, hereafter called the burnt velocity, may be evaluated by the following expression, where w_0 is the weight of the propellant and W is the weight of the inert parts of the round:

$$V_0 = \frac{g \int F dt}{w_0} \ln \left(1 + \frac{w_0}{W}\right). \tag{2}$$

For convenience, it is desirable to relate the thrust and the weight rate of burning of the propellant, dw/dt, by means of a term called the effective gas velocity, which is essentially constant for a given propellant burning in a particular rocket motor:

$$F = V_E \frac{dw}{dt}. \tag{3}$$

A combination of equations (2) and (3) results in the simplified expression:

$$V_0 = V_E \ln \left(1 + \frac{w_0}{W}\right). \tag{4}$$

[a] See Parts III and V for different treatments of these principles.

As a matter of interest, values of effective gas velocity for a number of the common rocket propellant combinations used in this country are presented in Table 1. These values are applicable only to the specific combinations of propellant and metal parts indicated.

One of the more effective measures of the overall efficiency of a rocket motor is the specific impulse, that is, the impulse available per unit total weight of propellant or of motor. In general, a well-designed rocket motor should yield an overall impulse of approximately 100 lb-sec per lb of motor; but not many of the rocket motors developed during World War II gave such high performance. Values of impulse per total unit weight of rocket motor for a number of the common rocket motors are included in Table 1. However, experimental work now in progress[b] under the cognizance of the Navy indicates that rocket motors can be developed with an overall specific impulse of approximately 120 lb-sec per lb. Such improvements result from careful revisions in design so as to decrease to a minimum the weight of the metal parts as compared to that of the propellant.

TABLE 1. Performance of JPN* propellant in several rocket motors.

Rocket motor	Grain†	Temp. (°F)	Effective gas velocity (fps)	Specific impulse (lb-sec per lb) Propellant $= V_{E/g}$	Rocket motor
2.25-in. Mk 10	Mk 1	10	6,670	207	32.6
		70	6,600	205	32.3
		130	6,350	197	31.0
3.25-in. Mk 6	Mk 13	0	6,180	193	53.5
		70	6,900	205	56.9
		140	6,420	202	56.0
5.0-in. Mk 1	Mk 18	−20	6,600	206	57.2
		0	6,680	207	57.4
		70	6,930	215	59.7
		140	7,000	218	60.5
		160	6,830	212	58.8

* See Table 2 for composition, etc.
† See Table 4 for ballistic characteristics.

[b] In summer of 1946.

[redacted]

5.2 PRACTICAL LIMITATIONS

The performance of rocket motors is limited by a number of practical considerations. The change in enthalpy upon reaction of the propellant does not in most instances exceed approximately 2,300 Btu per lb. Maximum specific impulse for a JPN propellant is presented in Figure 1 as a function of the reaction pressure and the expansion ratio of the nozzle. The performance indicated is the maximum that may be obtained; hence the effective impulse in actual operations is less, depending upon the loss of unburned propellant, frictional effects, and other causes.

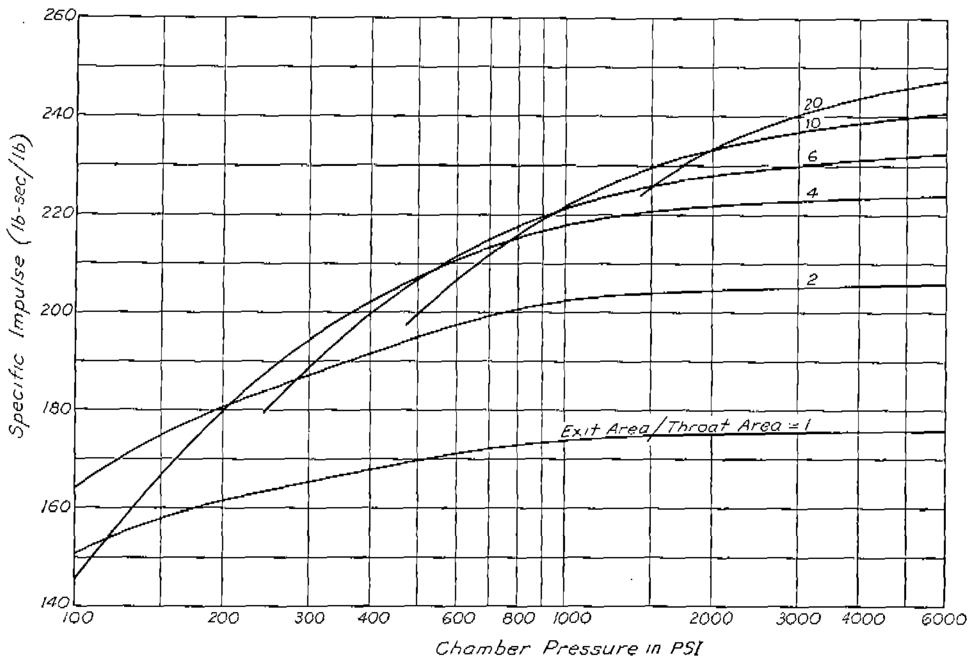

FIGURE 1. Possible specific impulse as function of reaction pressure and change in enthalpy.

Since the specific weight of most propellants is of the order of 100 lb per cu ft, the quantity of material to be placed in any given cross section of round is limited. Therefore, it is only possible to increase the quantity of propellant in a round of given cross section by increasing the length.

It is not possible, however, to increase the length indefinitely in the case of rockets with the nozzles located at a single section along the round, inasmuch as, when the burning occurs normal to the axis of the grain, frictional effects become of increasing importance as the round is lengthened. These limitations finally become controlling, and the reaction of the propellant becomes unstable, since the increase in pressure at the front end of the rocket motor over that obtaining at the nozzle causes a more rapid increase in burning rate than is compensated for by the increased flow arising from the higher pressure differential. In addition, relatively long grains fail as columns near the end of burning and either prevent the egress of gas from the motor, with a consequent failure of the metal parts, or yield large losses of unburned propellant, with a corresponding decrease in specific impulse.

These practical limitations can be overcome to a certain extent by the use of nozzles located at several sections along the axis of the rocket motor, but the resulting added complexity does not appear to justify this procedure except in a few special cases. Moreover, rounds which are excessively long in comparison to their diameter usually constitute a difficult handling problem. In general, it does not appear advantageous to utilize rocket motors whose length is much greater than 12 times the caliber of the round.

It may be of interest to note that liquid propellants do not impose the restrictions upon the geometry of the rocket motor that are encountered in the case of rockets with solid fuel. When liquid propellants are used, the reaction chamber may be made relatively small, and the fuel may be stored in containers of any shape suited to the exterior ballistic requirements of the round. The stowage of such artillery rockets, however, constitutes a problem that has not yet been solved. Nevertheless, it is believed that the use of liquid fuel in the larger artillery rockets is well worth consideration. The German government realized some success with liquid-fueled rockets, which in many instances exhibited superior ballistic characteristics to the solid-fueled rockets of comparable caliber. However, stowage difficulties were often encountered because of the corrosive action of the fuel.

5.3 BURNING CHARACTERISTICS OF PROPELLANTS

The colloidal double-base dry-extruded fuels used in artillery rockets burn upon the exposed surfaces at a weight rate which is roughly proportional to the area exposed. It is therefore of importance in the design of charges for artillery rockets to ensure that the change in burning area as the reaction proceeds is in accordance with the ballistic requirements imposed. For example, a marked change in the weight rate of reaction can be realized by relatively small changes in the cross section of the round. The reaction pressure also exerts a significant influence on the burning rate, as does the temperature of the propellant.

5.3.1 Influence of Position in Grain upon Burning Rate

It has been shown by numerous experiments that the burning rate of a solid propellant increases as the center of the web is approached. This is probably due in part to the gradual increase in the temperature of the unburned propellant because of thermal transfer and in part to the somewhat higher rate of transfer of radiant energy from the motor walls, the temperature of which rises during the latter part of burning. However, experimental measurements indicate that, even when the radiation and temperature effects described above have been eliminated, the burning rate of propellant under particular conditions of pressure and temperature is higher near the center of the web than near the original surfaces of the grain.[1] In Figure 2 is shown the influence of position upon burning rate for JP propellant, whose composition is given in Table 2, together with that of other typical propellants. It is apparent that the influence of position is significant in that the final burning rate for a propellant temperature of 0 F is higher than the initial burning rate for a propellant temperature of 70 F.

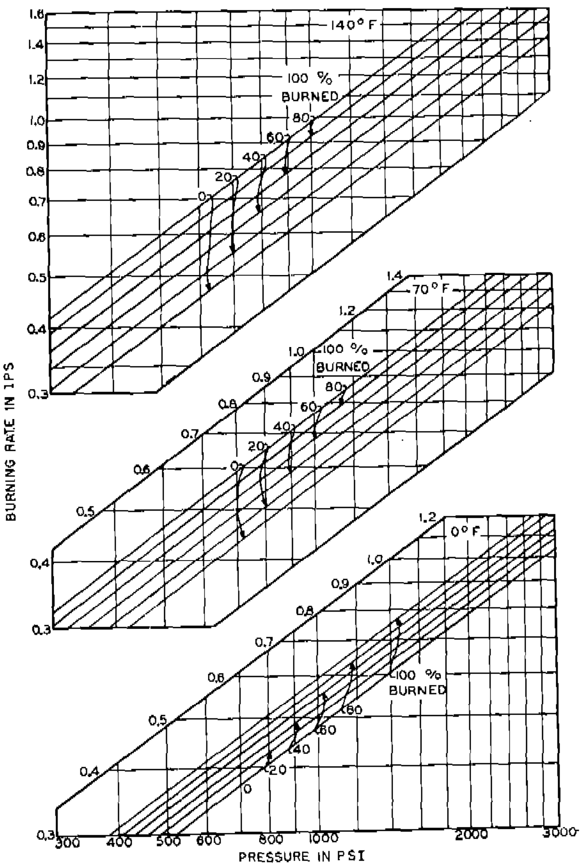

FIGURE 2. Instantaneous burning rates for JP propellant, showing influence of position.

5.3.2 Influence of Gas Velocity

There is, in addition, a significant effect which is directly related to the flow of the products of reaction past the reacting surface. In the case of relatively high weight rates of flow, the burning rate may be 30 or 40 per cent higher in the region of high gas velocity than where the reaction surface is surrounded by an essentially stagnant gas phase. This influence of erosion is shown in Figure 3, which presents comparative photographs of partially burned grains at the front and nozzle ends.

[redacted]

5.3.3 Influence of Pressure

For most double-base propellants, the burning rate is significantly influenced by the pressure of the reaction. This may in part result from the higher rate of energy transfer to the propellant from the products of reaction by radiation at the higher pressures. The influence of pressure may be approximated by an exponential equation of the general form

$$B = \beta\left(\frac{p'}{1{,}000}\right)^n, \tag{5}$$

where β and n are single-valued constants of the initial temperature of the propellant. The term $p'/1{,}000$ is used in place of the pressure in order to simplify numerical calculations and permit ready comparison of burning rates. It should be realized that at a reaction pressure of 1,000 psi abs the constant β represents the burning rate. Values of this burning rate at 1,000 psi for several common propellants are presented in Table 3. It should be emphasized that these values represent instantaneous burning rates at the *initial* propellant temperatures indicated. The table also records values of the exponent n.

5.3.4 Influence of Temperature

Temperature also influences the burning rate of a propellant.[1] The burning rate of the propellants which are markedly influenced by the reaction pressure are also particularly susceptible to the changes in the temperature of the charge. The importance of decreasing these effects is difficult to exaggerate since one of the primary attributes of a good rocket propellant is the small influence which reaction pressure and propellant temperature have upon the burning rate. Although most existing colloidal propellants exhibit relatively large variations in burning rate with temperature, it is evident from Table 3 that for certain of these the effect is much smaller than for the others. It is believed, therefore, that significant advancement in this direction can be made by careful investigation of the propellants which show the smaller influences of pressure and temperature upon burn-

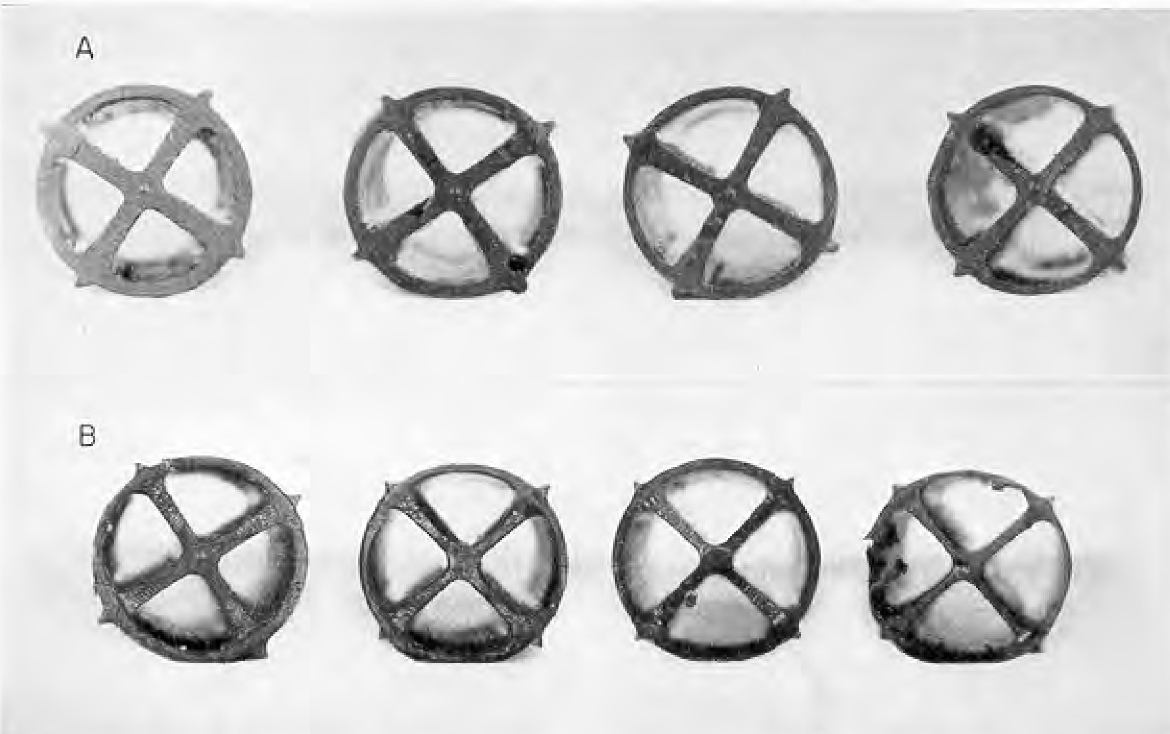

FIGURE 3. Partially burned grains showing appreciably less erosion at the front end (A) than at the nozzle end (B).

[redacted]L

TABLE 2. Composition and some thermal properties of typical rocket propellants.

Identification	Type	Composition: Constituent	Weight (per cent)	Heat of explosion (cal per gm)*	Adiabatic flame temperature (°F)†
JP	Ballistite of same composition as trench mortar sheet powder	Nitrocellulose (13.25 per cent N)	52.2	1230	5300
		Nitroglycerin	43.0		
		Diethylphthalate	3.0		
		Diphenylamine	0.6		
		Potassium nitrate	1.25		
		Nigrosine dye (added)	0.1		
JPN	Ballistite, modification of JP formula to improve stability	Nitrocellulose (13.25 per cent N)	51.5	1230	5300
		Nitroglycerin	43.0		
		Diethylphthalate	3.25		
		Ethyl centralite	1.0		
		Potassium sulfate	1.25		
		Carbon black (added)	0.2		
		Candelilla wax (added)	0.08		
JPH (FDAP 60)‡	Powder with burning properties similar to JPN, but of higher physical strength	Nitrocellulose (12.6 per cent N)	55.5	1260	5450
		Nitroglycerin	42.0		
		Ethyl centralite	1.0		
		Potassium sulfate	1.5		
		Carbon black (added)	0.2		
		Candelilla wax (added)	0.02		
Russian cordite (FDAP 44)‡	Double-base powder, cooler and slower burning than JPN	Nitrocellulose (12.2 per cent N)	56.5	880	3750
		Nitroglycerin	28.0		
		Dinitrotoluene	11.0		
		Ethyl centralite	4.5		
		Candelilla wax (added)	0.08		
H-4§	Double-base powder with burning rate intermediate between that of JPN and that of Russian cordite	Nitrocellulose (13.15 per cent N)	58.0	950	4000
		Nitroglycerin	30.0		
		Dinitrotoluene	2.5		
		Ethyl centralite	8.0		
		Potassium sulfate	1.5		
		Carbon black	0.02		
Ball powder	Compression-molded powder made by Western Cartridge Co.	Nitrocellulose (12.5 per cent N)	45.3	1050	4500
		Nitroglycerin	45.0		
		Trinitrotoluene	9.0		
		Ethyl centralite	0.7		
218B	Compression-molded composite propellant	Ammonium picrate	46.5	....	....
		Sodium nitrate	46.5		
		Butyl ureaformaldehyde resin	5.1		
		Plasticizer	1.5		
		Calcium stearate	0.4		

* Heat of explosion at constant volume with water in reaction products as liquid.

† Temperature with reaction at constant pressure.

‡ These numbers identify experimental lots of propellant manufactured by the Sunflower Ordnance Works, Lawrence, Kansas, that are representative of the designated types.

§ Designated T-2 by the Ordnance Department.

[redacted]

TABLE 3. Average burning rate data for typical propellants.

Powder	n*	β(ips)*			$\left(\frac{\partial \ln \beta}{\partial T}\right)_p$	$\left(\frac{\partial \ln p}{\partial T}\right)_{K_N}$ ††
		(0 F)	(70 F)	(140 F)	(1/°F)	(1/°F)
JP	0.71	0.551	0.671	0.815	0.0028	0.0096
JPN	0.69	0.564	0.651	0.752	0.0021	0.0068
H–4†	0.65	0.330	0.380	0.437	0.0020	0.0057
Russian (FDAP 44)	0.70	0.250	0.290	0.337	0.0021	0.0070
German	0.71	0.188	0.218	0.254	0.0022	0.0076
Japanese	0.42	0.278	0.311	0.349	0.0016	0.0028
Western Cartridge	0.64	0.340	0.393	0.454	0.0021	0.0058
218B composite	0.52	0.700	0.750	0.802	0.0010	0.0021
JPH	0.69	0.581	0.676	0.785	0.0022	0.0071

* Constants to use in relation $B = \beta\,(p'/1{,}000)^n$. † Designated T–2 by the Ordnance Department. †† K_N, nozzle coefficient, is the ratio of burning area to nozzle throat area.

ing rate; that is, the propellants of the H–4 type [c] rather than the JP or JPN types.

5.4 OPTIMUM PROPELLANT CHARACTERISTICS

Sufficient experience has now been accumulated to indicate the characteristics which are particularly desirable in a propellant for use in artillery rockets. It should be realized that these so-called optimum characteristics are from necessity somewhat general and that the importance of each of the several factors differs widely with various applications.

5.4.1 Burning Rate

It is desirable that the influence of pressure, temperature, radiation, and gas velocity upon burning rate be as small as is feasible.[1–3] Such a propellant can probably be approached by making suitable adjustments in composition and providing for adequate opaqueness. Modifications of composition are particularly efficacious with propellants of somewhat lower potential than the JP group—the H–4 stock, for example.

5.4.2 Physical Properties

The propellant should have adequate compressive strength and be resistant to impact, especially at low temperatures.[4–11] It appears that ultimate compressive strength is an index of the performance of colloidal propellants under conditions where great axial stress is applied to the grain during deflagration; and high impact values are important in the handling of rocket motors, especially at low temperature, since malperformance may result if the impact energies are not at least comparable to those realized with JPN propellant (approximately 12 ft-lb per sq in. at a temperature of 0 F). It is also necessary that the compressive strength not deteriorate unduly at the higher temperatures. For example, unsatisfactory field performance is obtained at 140 F with the Mk 13 grain in the 3.25-in. rocket motor Mk 6 when JP propellant, which has an ultimate compressive strength (at this temperature) of approximately 270 psi, is used.[8] However, satisfactory performance may be obtained with the same round at temperatures up to 150 F by using a propellant of identical ballistic characteristics but an ultimate compressive strength of 1,300 psi at 140 F.[7]

5.4.3 Stability

Although reasonable chemical stability is an important characteristic, most propellants involve compounds that tend to decompose with time. Nitrocellulose is especially troublesome in this regard, since its stability is significantly affected by manufacturing techniques and its rate of decomposition cannot usually be predicted with certainty. In order to improve colloidal double-base propellants from this standpoint, it will be necessary to investigate the characteristics of nitrocellulose, with particular attention to the influence of manufacturing

[c] Developed at Allegany Ballistics Laboratory for the 115-mm aircraft rocket. This propellant composition is designated T–2 by the Army Ordnance Department. See Chapter 13 of this volume.

[redacted]

techniques and the nature of the cellulose employed. In addition, the investigation should include suitable stabilizers; for the principal improvement of JPN over JP powder is in stability, which is apparently attributable to the substitution of ethyl centralite for diphenylamine.[d]

It is also important that the propellant be of such a physical nature as to be geometrically stable with respect to time. Any significant change in the geometry of the grain during stowage will result in a corresponding change in ballistic characteristics. These changes may be extensive enough to cause failure of the round.

5.4.4 Toxicity

It is important from a processing and loading standpoint that the propellant be as nontoxic as is compatible with satisfactory performance. Low toxicity is not a controlling requirement but is certainly a desirable attribute if the propellant is to be manufactured in large quantities with a minimum of special equipment and the fewest possible physiological difficulties for the operators.

5.4.5 Specific Impulse

The specific impulse of the propellant should be the highest that is feasible. In this respect, propellants now vary from approximately 100 to 220 lb-sec per lb; and it does not appear that many will be found in the near future for which the specific impulse will exceed the latter value.

5.5 INFLUENCE OF BURNING TIME ON TOTAL IMPULSE

From the standpoint of exterior ballistics, it is usually desirable that the burning time be as short as feasible, since for most types of rockets dispersion increases with burning time. However, this factor is not of great importance in connection with forward-firing fin-stabilized rockets launched from the exterior of aircraft, because the airstream induces an inherent stability in the rocket at the time of launching. The design of a rocket is therefore a compromise between the requirements which must be met in order to obtain low dispersion and high impact velocities at short ranges, and the limitations which are imposed by the design of the charge.

The weight of propellant per unit cross section of rocket is roughly a function of the burning time. The influence of burning time on the specific impulse per unit cross section of the round is shown in Figure 4 for JPN powder.[e] This relationship is not

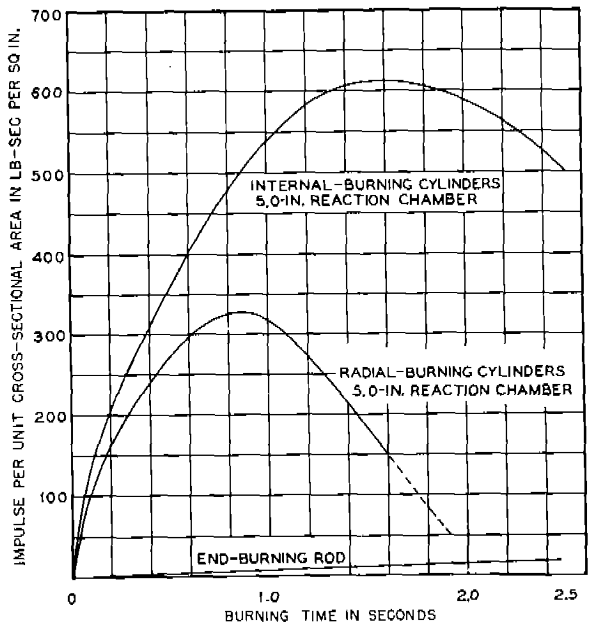

FIGURE 4. Impulse per unit cross-sectional area as a function of burning time for three types of charges.

strictly single-valued but covers a wide range, depending upon the particular grain section employed. Ultimately, the optimum behavior would be obtained with grain burning only on one end; but the burning time with existing propellants would be unduly long.

5.6 EFFECTS OF ACCELERATION

Acceleration imposes relatively large setback forces upon propellant grains. In the case of the

[d] For further information on stabilizers and their effects on propellant characteristics, see sections of the Division 8 Summary Technical Report covering the work of Pauling at CIT under Contract OEMsr-881, and reports submitted under that contract.

[e] The curves shown are based upon a burning rate of 0.65 ips, an internal area ratio (of burning area to ports area, i.e., the cross section available for gas flow) of 100 for tubular grains, and a motor of 5-in. inside diameter.

[redacted]

Mk 13 grain, for example,[11,12] a total force of approximately 340 lb is applied to the grid [f] by the grain during the early part of the acceleration of a round fired at 70 F, or about 525 lb for a round fired at 120 F. These forces cause elastic, and under some conditions plastic, deformation of the grain near the nozzle, with a corresponding decrease in the cross-sectional area through which the products of combustion flow from the forward end of the rocket motor to the nozzle. Since such port areas are relatively critical near the upper operating temperature limit of the round, relatively small changes in port area resulting from the elastic and plastic deformation of the grain may influence significantly the temperature at which unstable burning occurs.

Near the end of burning, the slenderness ratio of a grain becomes much larger; and, although the total force attributable to acceleration is smaller, the force per unit area resulting from the acceleration may nevertheless be enough to cause breakup of the grain in flight when practically no disintegration would occur under static conditions. If sufficiently extensive, the breakup of the grain will result in the failure of the round because of the marked increase in burning area. In any event, it will cause a distinct increase in pressure and the loss of unburned propellant.

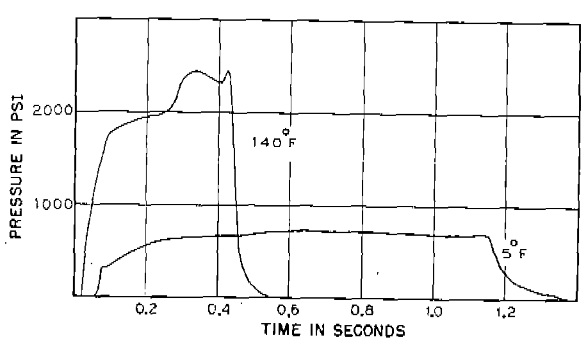

FIGURE 5. Pressure-time curves for Mk 13 grains showing breakup near end of burning at 140 F.

A significant part of the work of designing charges for rocket motors has involved studies of the influence of composition on physical characteristics [9,11,13] of propellants in order to decrease not only breakup near the end of burning but also deformation at the beginning. The quantitative nature of the breakup of grains is shown in Figure 5 and is described in some detail elsewhere.[12,14]

[f] A rocket component, usually of steel, which supports the rear end of the grain, and is supported by the nozzle.

5.7 TEMPERATURE LIMITS

The operating temperatures of rocket ordnance are greatly limited by the effects of temperature upon the physical and chemical characteristics of the propellant. At low temperatures the propellant becomes more brittle;[7,9] consequently, the grain may fail as the result of stresses imposed by acceleration or accidental impacts encountered in handling. Furthermore, at low temperatures the burning rate of the propellant decreases sufficiently for unstable burning to occur, during which the reaction substantially ceases and the propellant is reignited after an interruption of as long as a second or two. The reignition may be caused by contact with the hot metal parts of the rocket. In general, small grains begin to show unstable burning when the reaction pressure falls below 400 psi. However, in the case of large grains, where there are usually somewhat thicker gas films and where the energy loss per unit weight of propellant is somewhat smaller, stable reactions can be maintained at much lower pressures.

With respect to reaction pressure, the lower limit of stability depends upon the geometry of the particular charge under consideration.[15–18] At high temperatures the increase in burning rate introduces marked increases in pressure within the reaction chamber as a whole, and in many instances the upper limit of propellant temperature at which the round may be successfully fired is determined by the maintenance of stable burning near the end of the round opposite that at which the nozzles are located. Instability is not often encountered with rounds for which the ratio of burning area to port area is less than 100; however, the weight of propellant which may be stored in each unit cross-sectional area is limited. As a matter of interest, a number of the more pertinent interior ballistic characteristics of the several principal rocket charges developed by OEMsr-418 during World War II are recorded in Table 4.

For rounds in which the upper temperature limit is not controlled by the pressure developed during the reaction or by the occurrence of unstable burning of the propellant, it is limited by the physical properties of the propellant. The ultimate compressive strength decreases markedly with increase in temperature,[7,8,10] and in each case a temperature is reached at which the charge will not withstand the acceleration and frictional forces without under-

[redacted]

going significant plastic deformation. Under these circumstances the port area is decreased, and eventually a condition of unstable burning is reached.

TABLE 4. Ballistic characteristics of several rocket motor charges.

Grain	Mk 1*	Mk 13†	Mk 18‡
Burning area, A_c (sq in.)			
Initial	98.9	281.4	598
Final	66.4	260.0	613
Free port area, A_p (sq in.)			
Initial	0.96	2.54	6.3
Final	3.14	6.90	16.8
Ratio of burning area to port area, A_c/A_p			
Initial	103	110.7	105
Final	21	37.7	36.5
Average nozzle pressure (psi)			
−20 F	....	340	610
0	....	450	734
20	874	....	....
70	1,580	850	1,071
130	2,587	....	....
140	....	1,330	1,902

* In 2.25-in. Rocket Motor Mk 9.
† In 3.25-in. Rocket Motor Mk 7.
‡ In 5.0-in. Rocket Motor Mk 1.

characteristics of the metal parts may take place during the burning interval. In addition, there is a significant erosion of most nozzles, especially when the reaction pressure is high and the nozzle diameter small. This tends to increase the weight rate of flow for a fixed reaction pressure. Therefore, even if the

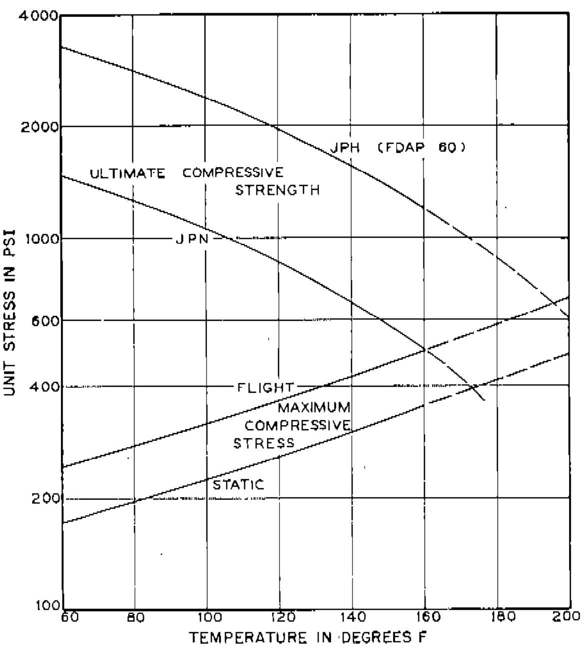

FIGURE 6. Influence of temperature upon the ultimate compressive strength of JPH and JPN propellants.

The influence of temperature on the ultimate compressive strength of two propellants is presented graphically in Figure 6, which also shows the corresponding stresses imposed during firing under both static and flight conditions for the 3.25-in. rocket motor with a Mk 13 grain.

5.8 CHARGE DESIGN

The design of propellant charges for rocket motors is controlled by the exterior ballistic requirements of the rocket and the deflagrating characteristics of the propellant. Since the object is usually to obtain a relatively uniform acceleration, which involves a constant weight rate of discharge from the nozzle during deflagration,[1,19–22] it is customary as a first approximation to design charges to burn neutrally. However, because of the thermal energy transferred to the metal parts of the motor and the consequent decrease in tensile strength of these parts, it may be necessary to arrange for the reaction pressure to decrease as the reaction proceeds.

This regressive type of charge design is particularly desirable in external-burning grains, such as the cruciform, where large changes in the physical charge is neutral in so far as its geometry is concerned, there will be a regression in the pressure as the reaction proceeds. For this reason it is often possible to design a neutral-burning grain and obtain the advantages of regressive burning by an increase in nozzle area from erosion. On the other hand, it is impossible to lengthen a particular rocket grain indefinitely without reaching an unstable situation wherein the increase in burning rate resulting from the rise in pressure is greater than the increase in rate of flow resulting from the same rise in pressure.

A number of typical grain sections employed in rockets developed during World War II or under investigation at that time are illustrated in Figure 7. All these charges burn externally, or both internally and externally, and hence require sufficiently heavy metal parts to withstand the reaction pressure at the end of burning, when the average temperature

[redacted]

of the metal parts is markedly higher than at the beginning. It appears that a significant decrease in the weight of the metal parts may be realized by utilizing an internal-burning grain of the type shown in Figure 8, which is inhibited on the periphery to prevent burning except in the axial perforation.[14,23-26] More recent experience with this

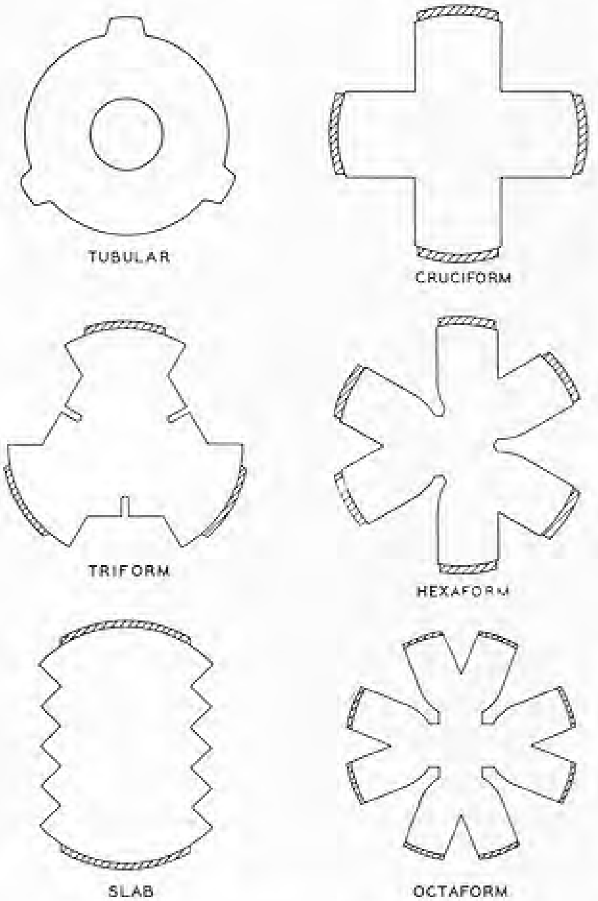

FIGURE 7. Cross sections of typical grains.

type of grain than was covered by the work of Section V of OEMsr-418 indicates that the surface temperature of the metal parts may be maintained below 140 F at all points except where the products of reaction come in contact with the interior of the wall. Such a grain is particularly adaptable for use in spin-stabilized rockets since it is well supported against centrifugal forces.

Internal-burning grains are considered one of the most promising means of increasing the performance of rocket motors, for it has been shown that their use will permit the overall specific impulse of the motor to be raised to a value well above 100 lb-sec per lb. It is possible to combine the internal-burning grain with the external-burning grain to obtain a relatively high density of loading and yet keep the burning time within the limits imposed by the exterior ballistic requirements for many types of rounds.[g]

The design of a solid fuel propulsion system for a particular application is based upon the fundamental principles of interior ballistics,[21,22,27] as well as upon an appraisal of the desired characteristics of the round. For example, if the round is to be one of maximum burnt velocity and straight trajectory, such as an antiaircraft rocket, it is necessary to

FIGURE 8. Internal-burning grain with cog-shaped axial perforation.

obtain the maximum impulse per unit of cross section. An end-burning grain of sufficiently rapid burning rate to give the desired acceleration and maintain a high terminal velocity would probably meet the requirements. However, since existing propellants do not even approach the necessary burning rate, an interior-burning grain must be employed.

The length of the grain which is to be used, and hence the average weight per unit cross section, is limited by the ratio of the burning area to the port area. This ratio is presented in Figure 9 as a function of the percentage of the cross-sectional area occupied by propellant for several ratios of the length to the diameter of an internal-burning grain with cylindrical axial perforation. This represents the minimum value of the ratio of burning area to

[g] This discussion does not include multiple-grain charges of the conventional type, such as were widely used with solvent-processed double-base propellants and in a number of foreign rockets. Such charges appear to have been effective in a number of applications; but they preclude insulating the motor wall from the transfer of thermal energy with the grain itself as can be done in the case of the internal-burning charge.

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port area that may be obtained with any shape of interior perforation and, therefore, is the optimum fraction of the cross-sectional area that may be occupied by propellant. However, the cylindrical cross section cannot usually be employed, since it

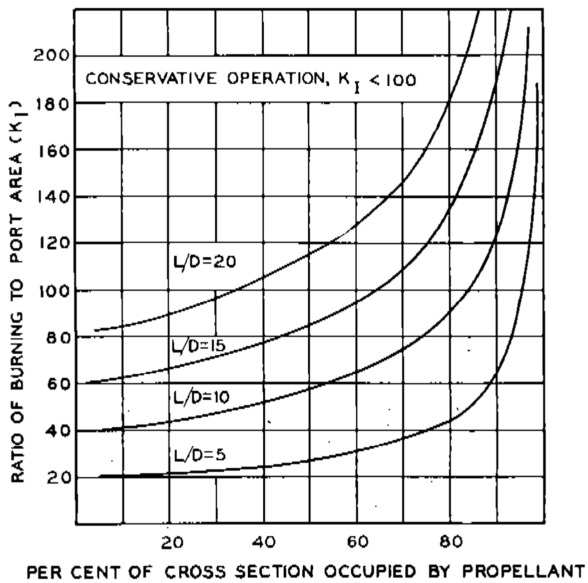


FIGURE 9. Influence of size of cylindrical perforation upon characteristics of an internal-burning grain.

results in an unduly progressive charge, except in instances where the web thickness is so small that an axial perforation with irregular periphery is not required.

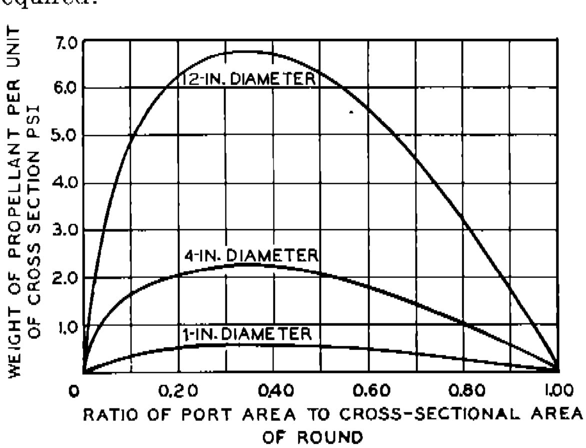


FIGURE 10. Influence of relative port area in an internal-burning grain upon loading density.

The data presented in Figure 9 permit the evaluation of the ratio of port area to cross-sectional area that should be employed in order to obtain the maximum weight of propellant in a particular rocket motor. In Figure 10 the weight of propellant that can be loaded into each unit cross-sectional area of a motor is shown as a function of the ratio of port area to cross-sectional area. In this instance it is assumed that the maximum acceptable ratio of nozzle port area to burning area is 100.

5.9 RECOMMENDATIONS

In the opinion of the writer, the use of internal-burning grains is the most promising approach to future developments in charge design and interior ballistics of rockets using solid fuels. In the case of short grains in which burning time is not important, a relatively small port area may be employed with a corresponding increase in the weight of propellant per unit of length and cross section. In situations where burning time is of importance, a propellant grain of the cross section illustrated in Figure 11

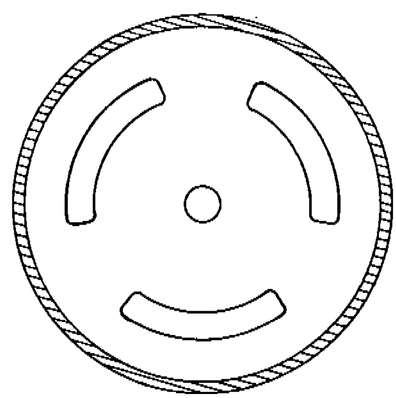


FIGURE 11. A shell-and-rod charge extruded as a single grain.

should prove useful. This grain is shown as extruded in a single piece, with the burning taking place on each of the exposed surfaces except the periphery. By appropriate modification of this design, it should be possible to obtain almost any desired burning time for a grain of given cross section.[h]

[h] The development of extrusion techniques and the details of the design of such grains have been carried out at the Naval Ordnance Test Station, Inyokern, subsequent to the termination of active work under OEMsr-418.

More specifically, the use of internal-burning grains in a number of applications appears desirable for the following reasons:

1. This type of grain avoids heat transfer to most of the wall of the rocket motor, thereby permitting the use of tubes of thinner steel, or possibly aluminum alloy, with a corresponding increase in the specific impulse of the motor as a whole.

2. The extrusion of concentric-web charges as a single unit will permit the production of internal-burning grains which are relatively simple to load and will withstand the high radial stresses associated with spin-stabilized rockets and still yield short burning times.

3. The internal-burning grain, or a variation thereof, permits nearly the optimum quantity of propellant per unit cross-sectional area that can be obtained with any geometric design yet proposed.

4. Present information indicates that internal-burning grains of JPN propellant burn stably over a relatively wide range of conditions and may be ignited without difficulty.

5. It is not necessary to provide a conventional grid for these grains.

6. The inhibiting of the exterior of the grains does not appear to constitute a production problem and may be accomplished by the application of cellulose acetate or ethyl cellulose as a spirally wrapped strip, a flat wrapped sheet, or a hot molded envelope.

It is believed that by the use of internal-burning grains rocket motors can be constructed to give overall specific impulses significantly in excess of 100 lb-sec per lb. Work should be directed toward the investigation of internal-burning charges which are closed at the end of the grain away from the nozzle, thus avoiding heat transfer to metal parts except in the immediate vicinity of the nozzle. In the case of long-range artillery rockets, this type of grain might be supplemented by an end-burning charge which would supply sufficient thrust to maintain high velocity after the end of burning of the primary charge. Such grains could be inhibited in a single piece. It should be emphasized, however, that these latter recommendations have not yet been investigated and hence should be considered only as proposals for future study.

At the present time the development of several types of internal-burning grains is in progress under the supervision of the Services. These should be useful in both spin- and fin-stabilized rockets. In applications where long burning time is permissible, a single axial perforation will probably suffice except in units of exceedingly large diameter.

5.10 LIQUID FUELS

As has been indicated, solid fuels have a number of limitations, notably the significant influence of temperature upon the ballistic and physical characteristics of the propellant. Moreover, the Germans had notable success with the use of liquid fuels in at least one large guided missile and in a limited number of simpler artillery rockets. It is believed, therefore, that the use of liquid fuels in large artillery rockets should be given careful consideration. Solid fuel may be used as a pressurizing agent, and the fuel containers need only be designed to withstand the reaction pressure at ambient temperature. The reaction chamber may be relatively light, and film cooling may be employed.

One of the primary requirements for a satisfactory liquid fuel for an artillery rocket is stability. At the present writing, binary liquid propellants seem to be more desirable for large artillery rockets than mono-liquid propellants. The probability of the detonation of a binary liquid propellant by small arms fire, or even high explosives, is small, whereas there will nearly always exist an energy threshold above which a mono-propellant will detonate. It appears that liquid-fueled rockets could be constructed in the larger sizes with a higher specific impulse for the rocket motor as a whole than the corresponding solid-fueled rockets. The cost of the metal parts may be somewhat higher; but, if the binary combination which is chosen shows adequate stability, the increase in performance would probably justify the added expense.

The transition from solid- to liquid-fueled rockets should be considered at calibers between 8 and 14 in., in so far as can now be determined. It does not seem practical to prepare single-grain solid fuel charges in diameters larger than perhaps 12 in. On the other hand, the use of liquid fuels in small rockets appears to be an unwarranted complication. The actual sizes and applications in which these two types of rockets will prove respectively superior remain to be established by development and Service experience. At the present time it is believed that the use of the oxides of nitrogen or nitric acid as the oxidant and aniline or one of its derivatives as the fuel is the most promising combination for the immediate development of liquid-fueled rockets. Hydrogen peroxide-hydrazene hydrate combinations do not appear well adapted to artillery rockets because of the difficulty of extended storage of hydrogen peroxide in sealed metal containers.

In conclusion it is reiterated that the development of liquid-fueled artillery rockets utilizing binary spontaneously ignitable liquid propellants appears to be worth while in spite of the added hazards involved, because of the marked simplification in the ignition system. This opinion is based upon satisfactory experience with colloidal propellants, which nearly always ignite if the case of the rocket motor is penetrated by gunfire.

[redacted]

Chapter 6

IGNITION

By *B. H. Sage*

6.1 GENERAL PRINCIPLES

Ignition in rockets has for the most part been satisfactorily accomplished with black powder igniters initiated by electric squibs, although in a number of instances percussion units have been employed. Other types of igniter charges have been investigated at least to some extent; but, of these, organic materials such as double-base propellants [1] have not proved particularly successful, and metal-oxidant mixtures,[2,3] although acceptable from the ballistic standpoint, offered no significant advantage over black powder in this respect and at the same time appeared to be somewhat more hazardous to handle. The mixture of this type which was found most satisfactory, magnesium powder and potassium perchlorate, is subject to detonation when fired in significant quantities; hence no extensive investigation was made of its detailed application. Moreover, since a large number of munition manufacturers are familiar with the methods of processing black powder, it is believed that the continuation of its use as an igniter charge in rockets fueled with double-base propellants is desirable. The present discussion will therefore be confined to black powder igniters and their characteristics.

In principle the ignition of a rocket motor utilizing a double-base powder as the propellant consists in transferring energy to the propellant at a sufficiently high rate to bring the immediate surface to the autoignition temperature, which is approximately 340 F. The detailed mechanism associated with this process is not well understood, although it appears that the igniters in question function primarily by the radiant transfer of energy from the products of reaction of the black powder to the propellant. Since the products of reaction of the black powder have a somewhat higher emissivity than those of the propellant, unusually high rates of burning of the propellant are obtained during the period that the products of reaction of the igniter are within the reaction chamber. This behavior is illustrated by Figure 1, which shows pressure as a function of time for the Mk 18 grain at several temperatures. It is apparent that the maximum ignition pressure changes only from 550 to 1,500 psi with a change in propellant temperature from −24 to 160 F. The corresponding change in reaction

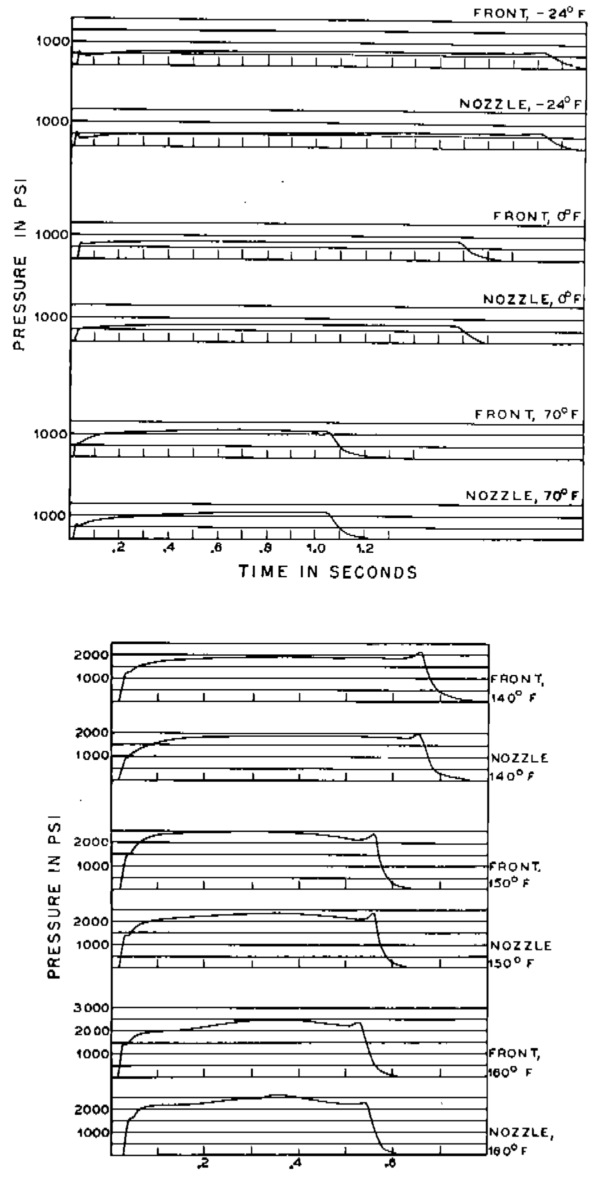


FIGURE 1. Pressure-time relationships for the Mk 18 grain.

pressure is from 570 to 2,500 psi. These somewhat typical data indicate that ignition pressure is not as greatly influenced as reaction pressure by the burning rate of the propellant.

An increase in the quantity of black powder increases the ignition pressure significantly. Within limits, an increase in the relative quantity of black powder per unit of free volume in the grain and igniter interval decreases the frequency of misfires or hangfires at temperatures close to the lower temperature limit of stable burning for the charge, although an increase in the size of the igniter beyond that necessary to produce an ignition pressure of approximately 1,000 psi does little to decrease the temperature at which reliable ignition can be obtained. However, an increase in igniter charge beyond this point or an increase in propellant temperature decreases the ignition delay, as is evident from Figure 2.

It should now be emphasized that the ignition pressure does not correspond to the pressure obtained when the igniter is fired in a free space of identical geometry involving only inert materials.

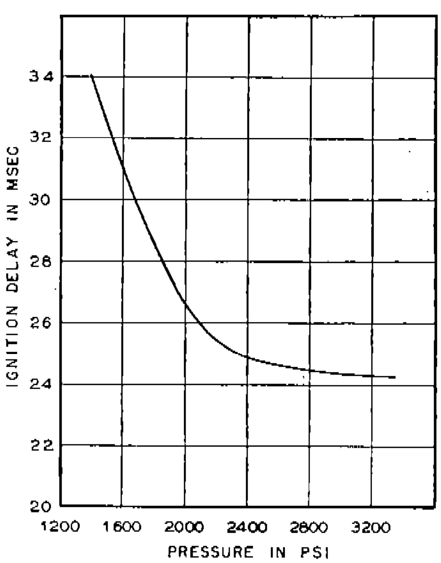


FIGURE 2. Relation between ignition pressure and ignition delay for 2.25-in. rocket motors.

In such instances the pressure within the chamber rises to perhaps 100 psi because of the reaction of the igniter alone. However, in combination with a grain of double-base propellant, the ignition pressure may be 1,500 psi. These values indicate the effect which the presence of the products of reaction of the igniter have upon the rate of reaction of the ballistite.

6.2 IGNITER CONSTRUCTION AND PERFORMANCE

A typical design of an igniter for a 2.25-in. rocket motor [4] is presented as Figure 3. The black powder is ignited by an electric squib. Experimental work

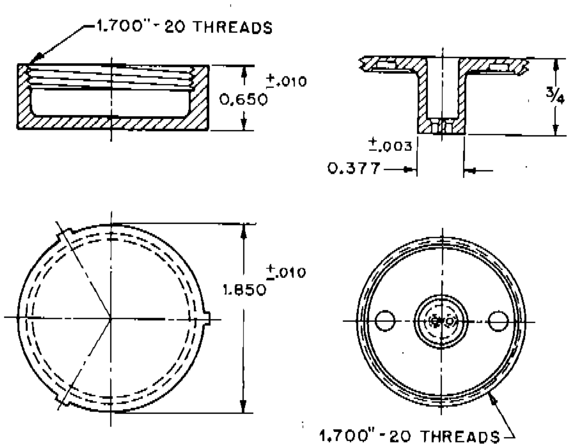


FIGURE 3. Plastic-case igniter for 2.25-in. rocket motor. This unit contains 12 g FFFG black powder.

has shown that the following is the approximate time schedule for the several steps in the ignition process. The values given represent the elapsed time in milliseconds from the application of the electric energy to the squib.

Melting of bridge wire	3 to 4
Initiation of black powder	5 to 6
Rupture of case	18 to 25
Ignition of propellant charge	25 to 36

It appears from this time schedule that the actual ignition of the propellant charge requires approximately one-third of the total ignition period and that the remainder is consumed in the action of the squib, the initiation of the reaction of the black powder, and the rupture of the case.

Black powder igniters are relatively cheap to prepare and involve materials that are readily available. In general, either a glazed or shell powder of approximately FFF granulation can be employed to advantage. It has been found that a decrease in the size of the particles to "dust" does not significantly decrease the ignition delay and often results in unsatisfactory performance because of the tendency of the dust to cake if slight quantities of moisture gain entrance to the igniter. On the other hand, efforts to sustain the ignition pressure by the use of coarse granulation did not prove particularly effective, and it appears that there is little to be gained by the use of a granulation coarser than that which will permit complete reaction of the black powder before expulsion from the rocket motor.

Since black powder is somewhat hygroscopic,[5] it is desirable to seal the igniter case in such a fashion as to prevent the entrance of moisture during storage. Small quantities of water up to approximately 1.5 weight per cent do not seriously affect the ignition characteristics (see Figure 4), but an increase in the water content of the black powder significantly above this value results in erratic and unpredictable

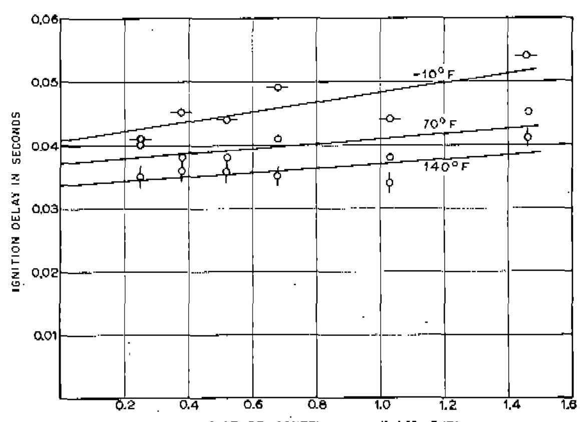


FIGURE 4. Influence of moisture in a black powder charge upon ignition delay for a 5.0-in. spin-stabilized rocket.

ignition delays and may cause disintegration of the active ingredients of the squib. A water content of 1.5 per cent corresponds to equilibrium at a relative humidity of about 92 per cent at 80 F.

The electric squibs employed in most of the igniters with which this group has been concerned were of a standard deflagrating type prepared by a commercial munitions manufacturer. The current required was approximately 0.5 ampere in order to cause the bridge wire to fail in 3 or 4 milliseconds. If currents significantly less than 0.5 ampere were used, the time required for the failure of the bridge wire was uncertain and increased rapidly until it exceeded 1 second with currents of approximately 0.2 ampere; but increasing the current above approximately 1 ampere did not significantly affect performance. Squibs can be prepared requiring much smaller energies than those indicated above; for example, experimental squibs have been tested which give reproducible ignition delays with energy requirements of less than 20 ergs.

The squibs employed in many of the rockets developed by this group during World War II were susceptible to ignition by high-voltage electric discharge. It was found that the voltage applied between the face of the squib and one of the leads differed markedly from unit to unit, apparently because of irregularity in the depth to which the bridge wire was immersed in the active ingredients, and that normal statistical variation resulted in a limited number of squibs which may have been sensitive to the static discharges likely to be encountered in handling. However, in the course of loading several hundred thousand squibs, only two ignitions occurred which may be attributed to static discharge.

The squibs were fired by means of a low-voltage electric circuit, part of which was located within the rocket motor. The connection between the interior of the rocket motor and the leads to the firing circuit was accomplished in a number of ways, depending upon the design of the particular motor; but the maintenance of an adequate seal to prevent the entrance of moisture was troublesome. It may therefore be desirable in the future to consider the use of low-energy squibs and induction firing [6] in order to avoid the necessity of sealing the leads and, particularly in the case of rockets fired from automatic launchers, connecting the rounds to the firing circuit. The possible hazards arising from stray electromagnetic fields may be minimized by the use of specially wound coils requiring unusual configurations of field in order to induce the requisite energy in the interior circuit.

Because of the relatively fragile nature of the squib and the black powder grains, it is customary to assemble the igniter in some kind of semirigid container. From the standpoint of short ignition delay it is probably desirable to maintain the ratio of the surface of the container to the volume of the container as small as possible, but small digressions from the spherical shape which is thus indicated do not materially influence performance. Igniter cases have usually been prepared from plastics [4,7,8] and metal. Igniters with tin plate cases [9] have proved to be entirely satisfactory with motors having nozzles large enough not to be plugged by fragments of the case; a typical design for use with a 5.0-in. rocket motor is shown in Figure 5. Diffusion of

[redacted]

nitroglycerin from the double-base propellant causes deterioration of plastic cases; but, under normal conditions of storage in Service use, the ballistic performance of the igniters does not seem to be modified significantly.

Some type of very thin metal igniter case of cylindrical shape would appear to be satisfactory for internal-burning grains, and it is possible that an alloy of relatively low melting point might be

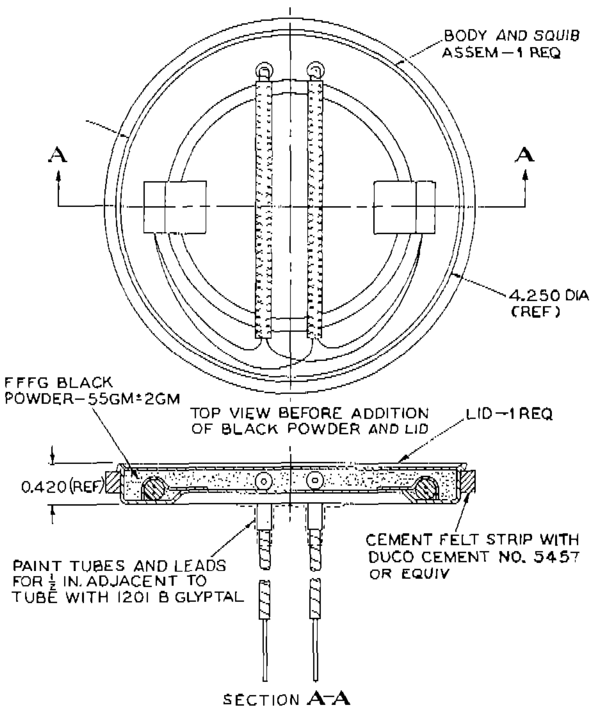


FIGURE 5. General arrangement of Mk 14 igniter for a 5.0-in. rocket motor.

desirable to avoid the difficulties associated with nozzle plugging. To facilitate loading and prevent movement of the squib with respect to the case during vibration, a small stamping or other piece should be provided to hold the squib in place. In general, the heavier the wall of the igniter case, the greater the stresses imposed upon the grain at the time of the rupture of the case; however, up to a certain point an increase in the weight of the case decreases and renders more reproducible the ignition delay. Under certain circumstances cloth bag igniters appear to deteriorate more rapidly when subjected to vibration than do either the metal or plastic units.

6.3 SUMMARY OF REQUIREMENTS

For optimum performance an igniter should initiate the reaction of a propellant charge in a minimum of time. Apparently the time required to initiate the reaction of double-base propellant is from 6 to 10 milliseconds, and this time is roughly independent of the quantity of black powder employed. Normal igniters of the present types usually give ignition delays from 25 to 36 milliseconds, depending upon the geometry of the rocket motor and the design of the igniter. It is doubtful whether the reaction of the propellant charge of a rocket motor with a single igniter can be initiated in much less than 12 milliseconds; and the decrease of ignition delay to this value must be accomplished for the most part within the igniter itself.

An igniter should not react with sufficient violence to place undue stresses upon the propellant charge. For this reason it is desirable to make the case of the igniter no heavier than is necessary to confine the ignition charge until it is ignited. Approximately 6-mil tin plate appears heavy enough to meet this requirement.

An igniter should function over a range of temperatures which corresponds to the range of successful operation of the round as a whole and should also be reasonably resistant to the influx of moisture. These, as well as the other requirements summarized above, can be satisfied with either plastic case or metal case igniters of suitable design.

[redacted]

Chapter 7

DRY-PROCESSED DOUBLE-BASE PROPELLANTS

By *B. H. Sage*

7.1 CLASSES OF PROPELLANTS[a]

THE NOMENCLATURE associated with the designation of the several types of double-base propellants is not entirely clear. For present purposes they will be considered in two general classes: those which are processed by the use of solvents, and those which are processed from mixtures with water to the finished propellant without the use of solvents. The first class will be referred to as solvent-processed propellants and the second as dry-processed propellants. Although the cost of manufacturing propellants by either of the two methods is comparable, the removal of solvent from grains having a web thickness greater than 0.5 in. requires such unusually long periods of time and dimensional uniformity decreases to such an extent that grains with thick webs are usually processed by the dry method. The dry processing probably involves a slightly greater hazard during manufacture but yields a product of good dimensional uniformity which may be prepared in web thicknesses limited only by the scale of the available extrusion equipment.

7.2 COMMENTS ON MANUFACTURING METHODS

No effort will be made in this report to discuss the relative merits of the several methods of preparing dry-processed propellants, but a few general comments appear to be in order. Although conventional methods are used in the manufacture of the requisite nitroglycerin and nitrocellulose, it has been found that the nitration and source of the cellulose influence significantly the physical characteristics as well as the ballistic potential of the propellant. Double-base powders prepared from nitrocellulose made from wood pulp are much more difficult to extrude than powders of identical composition prepared from nitrocellulose made from cotton linters. For this reason most of the nitrocellulose employed in the manufacture of dry-processed double-base propellant in this country during World War II was prepared from cotton linters. In so far as is known to the writer, the reasons underlying this difference in extrusion characteristics are not yet clear. However, it is evident that nitrocellulose with a wood pulp base yields a powder which tends to check and crack upon extrusion and which gives a much higher velocity distribution across the die than powder derived from nitrocellulose with a linters base.

In the case of the slurry process, the nitrocellulose is mixed with a relatively large quantity of water and agitated. The nitroglycerin is then introduced, together with certain of the additive ingredients, and the whole permitted to come to substantial equilibrium. The nitroglycerin is assimilated by the nitrocellulose. The resulting solid or plastic phase is separated from the water by means of centrifuges. At this point in the process the paste contains approximately 30 per cent water by weight. It is allowed to age and dry in bags, where the moisture content is reduced to approximately 6 per cent. After blending, the material, which is now called "dry paste," is placed upon differential-speed rolls of a design adapted from the rubber industry and rolled sufficiently to colloid the stock reasonably well. It is then removed from the differential-speed rolls and transferred to even-speed rolls, where further mechanical energy is added in the course of a number of "bookfolding" operations. The resulting sheet is approximately 0.050 in. thick and slightly translucent, although the addition of approximately 0.2 per cent carbon black renders it relatively opaque.

The details of the manufacturing[b] of double-base dry-processed propellant varied significantly from plant to plant in accordance with the availability of facilities; nevertheless, there appeared to be no marked variation in the quality of the product. A relatively large number of fires occurred in the course of the rolling operation. However, the use of special deluge equipment reduced the number of

[a] See also Part III.

[b] All under Ordnance Department contracts.

injuries to personnel to a relatively low value. Detonation of the propellant stock on the rolls has been known to occur.

7.3 EXTRUSION OF DOUBLE-BASE STOCK

The extrusion of dry-processed double-base powder was first carried out in the late fall of 1941.[1] Additional work was done on a somewhat larger scale shortly thereafter,[2] and relatively large grains of JP, JPN, and JPH (see Table 2 of Chapter 5) sheet stock into finished grains.

The extrusion operation involves the heating of the sheet powder to a temperature of from 100 to 140 F, depending upon the grain section to be prepared, and the insertion of the sheet stock as a "carpet roll" or as flakes into a horizontal or vertical press. After the press has been closed and the pressure lowered to approximately the vapor pressure of water at the charge temperature, the volume of the charge is reduced until the charge is extruded at pressures from 4,000 to 9,000 psi.

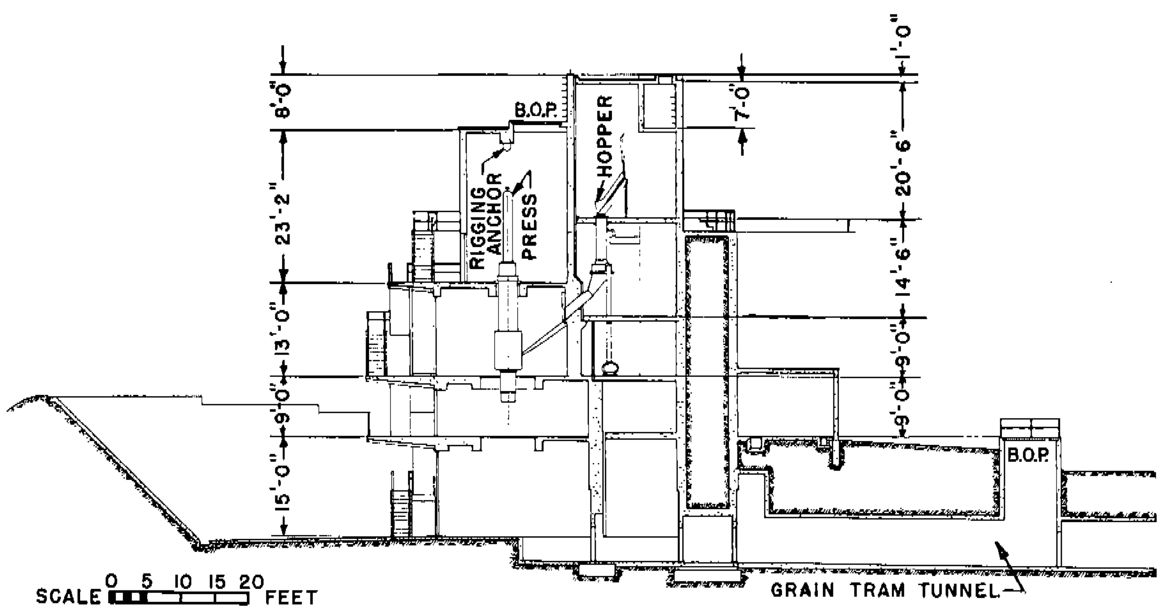


FIGURE 1. General arrangement of 18-in. vertical extrusion press at Naval Ordnance Test Station, Inyokern.

were extruded at a somewhat later date.[3] A small-scale extrusion plant was designed for the Navy Department.[4] This was built and operated by the Navy at the Naval Powder Factory, Indian Head, Maryland. The methods of preparing more complex multiweb grains [5] are not particularly difficult, and there are indications that conventional die design as practiced by the plastics industry may be employed in the extrusion of a number of the double-base dry-processed propellants. A description of the experimental production facilities developed in the Pasadena area by CIT under OEMsr-418 is available.[6,7] These facilities [c] were used mainly for the processing

Commercial manufacturers[d] throughout the country utilized horizontal extrusion presses varying in diameter from 8 to 15 in. But such presses are difficult to feed with other than "carpet roll" extrusion charges; hence nearly all material to be reworked in the commercial establishments was rerolled into sheet stock and in most instances blended with a certain amount of new "dry paste." On the other hand, the group at the California Institute has generally favored the use of vertical presses because they permit the direct extrusion of rework material without an intermediate rolling step. Charges of double-base stock which had been cut into relatively small pieces were fed to the vertical presses

[c] Most of this equipment has been moved to the Naval Ordnance Test Station, Inyokern, California, and to Picatinny Arsenal.

[d] All under Ordnance Department contracts.

[redacted]

without difficulty. However, the vertical presses required more extensive barricades [8,9] than would have been necessary for horizontal presses. It is probable that each type of hydraulic press has its advantages and shortcomings for a particular situation.

Although it does not appear desirable to enter into a detailed discussion of the design and operation of extrusion presses, a schematic drawing and a photograph of an 18-in. vertical extrusion press are presented as Figures 1 and 2. This press, which is located at the China Lake Pilot Plant of the Naval Ordnance Test Station, Inyokern, California, was designed and the equipment constructed by Section V of OEMsr-418. The installation requires relatively large barricades in order to permit operation with propellants of marginal compositions for which the frequency of ignition during extrusion would be unusually high.

Double-base dry-processed propellant can be extruded from dies made up of conical and cylindrical sections, or from more complex configurations which give lower rates of shear within the propellant for a given extrusion velocity. A die used in a 12-in. vertical extrusion press [8] for the preparation of the Mk 13 grain is illustrated in Figures 3 and 4. Since the Mk 13 grain is of cruciform section, no "stake" is required; one is necessary, however, with a die for an axially perforated grain. It is essential that the dies be prepared with a relatively high polish in order to decrease the friction between the metal surface and the propellant being extruded. This avoids localized excessive temperatures at the interface, which have probably caused at least one press ignition.[10] A low coefficient of friction also decreases the velocity distribution within the propellant during extrusion and consequently may reduce the frequency of inhomogeneities in the extruded product.

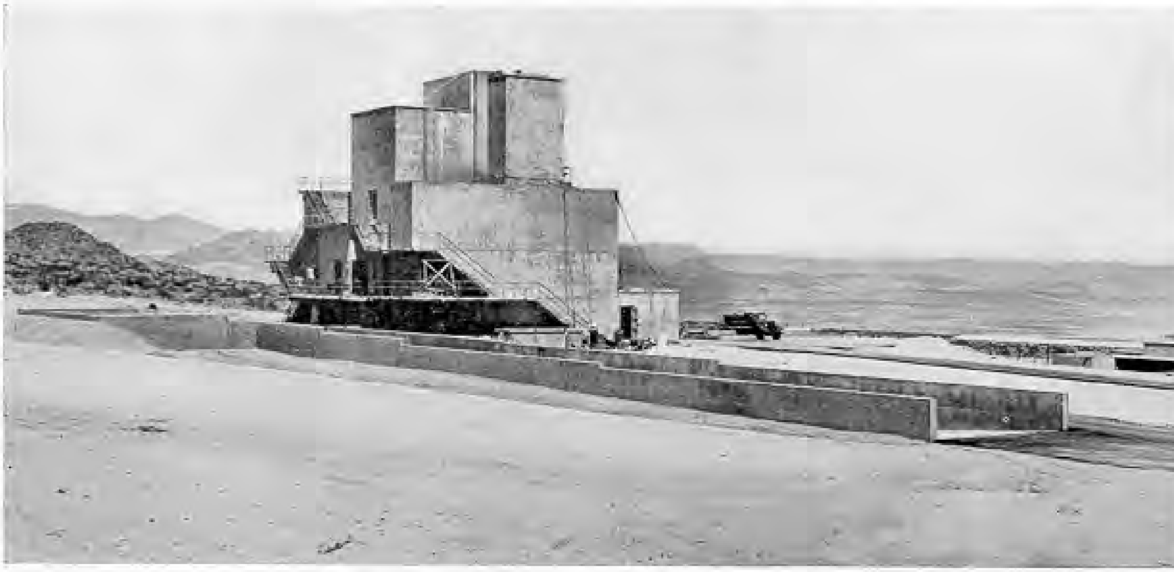


FIGURE 2. Eighteen-inch vertical extrusion press at Naval Ordnance Test Station, Inyokern.

The extruded grains immediately undergo a significant change in size, which is usually a dimensional increase with respect to the die and which in some cases may amount to as much as 8 per cent. Furthermore, relatively high stresses remain which are relieved but slowly at room temperature. Therefore, a grain that has not been annealed will gradually shorten and become larger in cross section. In order to avoid this difficulty, which may exert a significant influence on the upper safe-operating limit of the charge, the grains are annealed in a low-velocity airstream held at 140 F for a period of approximately 4 hours per inch of web thickness. During this period the grains are placed upon racks which assist in eliminating any unusual longitudinal curvature. An extruded dry-processed double-base propellant grain so supported as not to be deformed by its own weight will be straight within approximately 0.05 in. per ft of length.

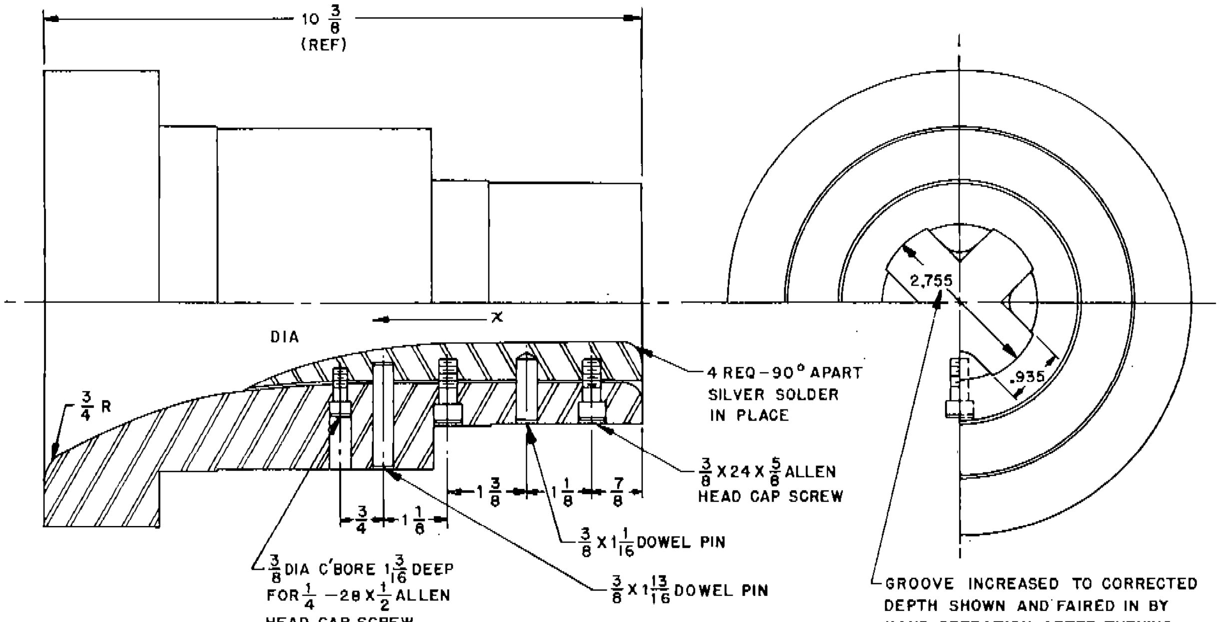
[redacted]

7.4 MACHINING

After being extruded and annealed, the grain is subjected to such machining operations as may be requisite. In general, it is not necessary to machine the periphery of the grain, since in this respect it is possible during extrusion to hold the dimensions within the limits imposed by ballistic requirements. However, it is usually necessary to bring the grain to a given length and weight within relatively small tolerances. Furthermore, it is often desirable to apply a plastic support to the end of the grain to aid in the distribution of the setback and friction forces over its cross section. In order that the cellulose acetate or ethyl cellulose reinforcement may be bonded satisfactorily, the surfaces of the propellant and the plastic must match closely. For this reason the grain is usually faced or sawed rather than cut.

Most double-base propellants can be machined readily with conventional machine tools at much higher tool speeds and feed rates than are employed for metals. Some success has been realized in the use of plastic saws, but these give a chip with a somewhat higher specific surface than is obtained by turning or milling. Fires during machining operations are relatively rare and can usually be traced to foreign material in the ballistite, exceedingly dull tools, or the inadvertent relative motion of metal and propellant surfaces which are in contact.

For the most part, the weight of a propellant grain can be held within the desired limits by appropriate control of its cross section and length. It is usually possible to machine grains to a fixed length or into groups of fixed length determined by grading the several grains according to their cross sections. The specific weight of extruded double-base propellant is remarkably constant for a given composition; in fact, for JPN powder it is generally within 0.5 per cent of 100.5 lb per cu ft.



X DIST	DIA	X DIST	DIA
0.000	2.710	4.000	3.398
0.400	2.717	4.400	3.546
0.800	2.737	4.800	3.716
1.200	2.772	5.200	3.913
1.600	2.820	5.600	4.144
2.000	2.882	6.000	4.423
2.400	2.957	6.400	4.766
2.800	3.047	6.575	
3.200	3.150		
3.600	3.267		

FIGURE 3. General arrangement of die used in extrusion of Mk 13 grain.

7.5 INHIBITING

The change in burning area as the reaction progresses can be controlled within limits by modifying the cross section of the grain. However, in the case of a simple external-burning grain such as the cruciform section, which is typified by the Mk 13 and the Mk 18 grains, it is not possible to obtain a neutral or progressive burning surface without preventing or inhibiting burning at certain points on the surface of the grain. It has been found that it is a relatively simple matter to prevent the surface reaction of double-base propellant by the application of suitable coatings. The application of strips or sheets of cellulose acetate was the method commonly used in this country for inhibiting.

FIGURE 4A. Die used in extrusion of Mk 13 grain.

In the case of the cruciform grain, the arms were inhibited on the periphery, as shown in Figure 5 for the Mk 13 grain. In general, the thickness of the inhibitor was increased with the web thickness and was approximately 0.10 in. for the Mk 13 grain. The cellulose acetate inhibitor strip was prepared by extrusion and applied by the use of solvents miscible with both propellant and strip. Numerous solvents are suitable; and, as a matter of convenience, mixtures of Cellosolve and methyl Cellosolve (2-ethoxy- and 2-methoxy-ethanol) were employed. The relative quantities of these compounds were varied, depending upon the temperature of propellant at the time of application, in order to obtain the desired rate of softening of the cellulose acetate. The strips were applied manually to most of the grains used in Service rockets. Efforts have been made to develop automatic equipment for this purpose, since undesirable physiological effects are usually experienced by operating personnel as a result of either the solvents themselves or the nitroglycerin in the propellant. However, such equipment has not yet proved entirely satisfactory.

The British used a slightly softer material consisting of cellulose acetate and triacetin, i.e., dummy cordite. It is believed that the development of a suitable plastic of very low elastic limit might permit the use of automatic machines for the application of the inhibitor strips and still avoid the difficulties which otherwise result from lack of uniformity in the curvature of the grains and strips. However, the inhibitor strip on the completed grain must be hard enough not to be unduly deformed during normal handling and storage.

In the case of end-burning or internal-burning grains where the entire periphery is inhibited, two techniques appear to be promising. One involves wrapping the grain with relatively thin cellulose acetate, or perhaps other plastic sheet, using appropriate plasticizers to obtain a satisfactory bond between the propellant and the sheet. This method may be modified to permit thin plastic tape to be wrapped spirally on the grain. Experience with the latter form of inhibitor indicates that larger geometric irregularities may be permitted than can be tolerated with full-width sheet. The second approach involves extruded or molded tubing which is shrunk or molded onto the grain; but, in so far as the writer is aware, these techniques, which were developed by the British, have not been widely used in this country.[a] The end of an inhibited internal-burning cylindrical grain is shown in Figure 8 of Chapter 5.

[a] The experience of the Allegany Ballistics Laboratory with these methods of inhibiting is indicated in Section 11.2.1 of this volume.

7.6 CHEMICAL STABILITY

The stability of propellants is of importance in connection with their storage. Of the principal components, it is probably the nitrocellulose which contributes most to the chemical instability of multibase propellants. Stabilizers are therefore employed to avoid the accumulation of the oxides of nitrogen which results from the spontaneous decomposition of the nitrocellulose, since the presence of these free oxides of nitrogen apparently accelerates further decomposition.

In JP propellant, which was developed primarily for use with trench mortars, the stabilizer was diphenylamine. However, it is now evident that this material is unduly active and tends to accelerate the decomposition of the nitrocellulose. Of the relatively large number of stabilizers that are available, ethyl centralite seems to be the optimum for the stabilization of colloidal propellants with a nitroglycerin-nitrocellulose base. This material can be incorporated readily into the propellant, and the equilibrium at constant pressure is such that the oxides of nitrogen do not contribute to any significant extent to the decomposition of the nitrocellulose. It is probably desirable to continue the investigation of stabilizers; but it is believed that studies of the characteristics of nitrocellulose will contribute equally, if not more, to the stability of double-base propellants, for it is apparently the combination of nitrocellulose and stabilizer which results in a propellant with a long storage life under adverse conditions, rather than the choice of a particular stabilizer.

Thick-webbed grains of dry-extruded double-base propellant involve a somewhat unique problem with regard to stability, since it appears that small quantities of the gases from the decomposition of the nitrocellulose do not react completely with the stabilizer. In the case of thin-webbed grain, these gases diffuse to the exterior surface and cause no particular difficulty. With thick-webbed grains, however, the fugacity of this material within the grain may reach relatively large values. The corresponding mechanical stresses result in cracking of the grain, usually parallel to the axis, or spalling on the surface. Since the occurrence of such defects during storage may well become a factor limiting the size of large propellant grains, it is believed that improvement in the effectiveness of stabilizers will result in an increase in the size of grain which it will be feasible to manufacture and store.

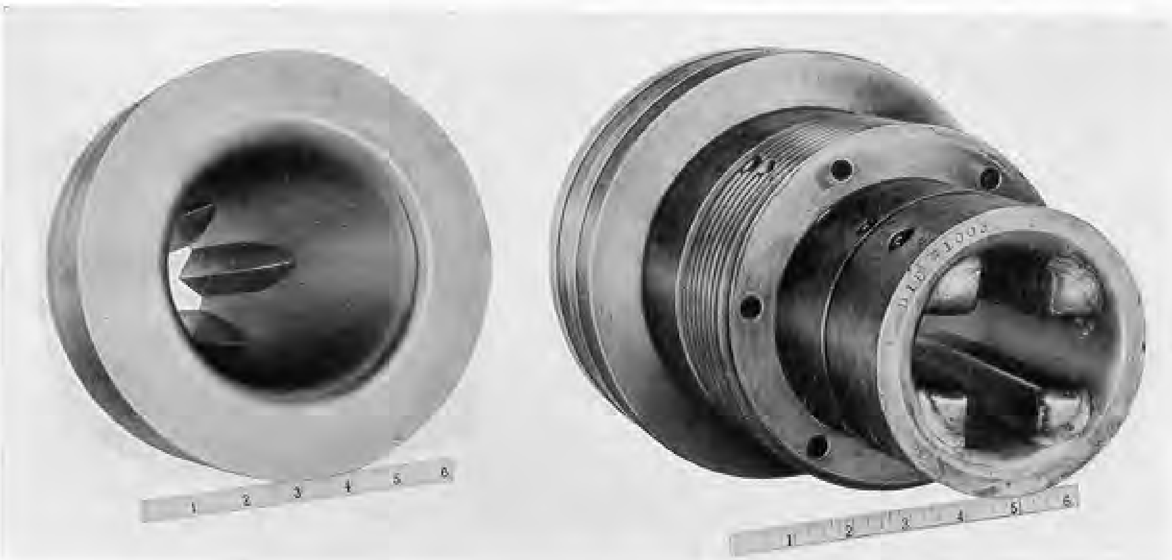


FIGURE 4B. Die used in extrusion of Mk 13 grain.

7.7 STABILITY OF BURNING

In the study of the deflagrating characteristics of propellants it has been found that irregular reaction pressures are often encountered, especially with propellant of relatively high burning rate, such as JPN or JPH. When efforts were made to utilize charges of these materials which were permitted to burn from extended plane surfaces, the reaction was sufficiently erratic to cause mechanical failure of the grain. For this reason it was impossible to use single continuous strips of inhibitor on the periphery of the arms of the Mk 13 and Mk 18 grains. Moreover, attempts to obtain regular burning within a cylindrical annular perforation also resulted in mechanical failure of the grain because of the instability of the reaction. This situation was overcome by drilling radial holes at somewhat random intervals but spaced longitudinally not more than 1 in. apart. It has recently been found that relatively stable burning occurs in a star-shaped perforation.

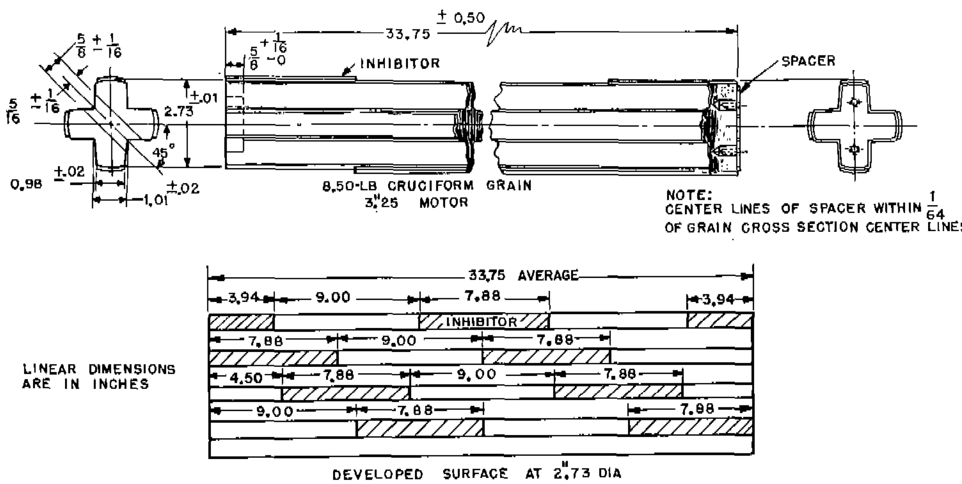


FIGURE 5. General arrangement of inhibitors on Mk 13 grain.

With propellants of intermediate burning rate (0.4 ips at 70 F and 1,000 psi), of which British cordite is typical, it is possible to obtain stable burning with less frequent interruption of plane surfaces, or surfaces of relatively large radius of curvature, than is necessary with propellants having high burning rates (0.65 ips at 70 F and 1,000 psi, for example). In the case of propellants with burning rates of less than approximately 0.25 ips at 70 F and 1,000 psi it has been found that stable burning may be obtained with almost any shape of grain that does not involve excessive energy exchanges associated with friction. (Instability attributable to frictional effects is entirely separate from the type under discussion.)

The mechanism of unstable burning not directly associated with frictional effects is not thoroughly understood, but may be related to resonance. A sufficient background of empirical information is available, however, to permit the design of propellant charges of each of the powders commonly employed.

7.8 SUMMARY AND RECOMMENDATIONS[f]

The general status of the knowledge relating to dry-processed double-base colloidal propellants that have been employed in artillery rockets has been indicated in the foregoing discussion. It now appears that rocket motors having an overall specific impulse of approximately 110 lb-sec per lb can be developed which will operate at temperatures between −30 and 130 F. The burnt velocity obtainable with such a rocket motor depends almost entirely upon the payload to be carried. Five-inch rounds of reasonable length-to-caliber ratio can be made with burnt velocities greater than 3,500 fps and with payloads of approximately 10 lb. However, if the payload is increased until it is equivalent to a shell of comparable caliber, velocities in excess of 2,500 fps are unlikely. The use of internal-burning grains prepared from existing propellants seems feasible and not too costly. Such grains permit rates of spin in excess of 400 rps to be obtained at temperatures up to 120 F without failure of the grain.

Regarding propellants with potentials in excess of 200 lb-sec per lb, there is little to be gained at present by modifying the composition greatly from

[f] See Chapter 13 for additional recommendations.

[redacted]

that of JPN powder. Such propellants apparently have insufficient latitude to permit the addition of buffer components which will decrease the influence of temperature and pressure on reaction rate. However, propellants having potentials of the order of 150 lb-sec per lb are promising in this respect; and it is probable that hydrocellulose and magnesium oxide in conjunction with potassium nitrate will prove particularly useful. It is believed that, for the time being, developments requiring propellants of intermediate potential may proceed satisfactorily on the basis of material approximating the H–4 composition recorded in Table 2 of Chapter 5. It does not appear that any new propellant which would justify delaying the program will be available within the next year (1947) in sufficient quantities for experimental production. Accordingly, it is recommended that the development of rocket ordnance involving dry-processed double-base colloidal propellants utilize the existing JPN formulation for a high-potential, fast-burning powder and the H–4 formulation which can be dry-extruded for a powder of intermediate potential.

Two lines of endeavor should probably be followed in the further development of dry-processed colloidal propellants. In the first place, a careful investigation should be made of the so-called buffer constituents which appear by their control of the chemical equilibrium to decrease the influence of temperature and pressure on the burning rate. Particular emphasis should be given the application of these constituents to the propellants of higher potential, with which they do not now appear to be sufficiently effective to warrant their use. Such studies, together with investigations of stabilizers and the character of the nitrocellulose, can well be carried out at academic institutions or government laboratories. The second approach should involve a systematic study of the influence of composition upon the physical, chemical, and ballistic characteristics of a number of systems comprising the principal components of existing double-base propellants. In this connection it is believed that investigation of such restricted ternary systems as the nitrocellulose-nitroglycerin-ethyl centralite system and the ethylene glycol dinitrate-nitrocellulose-ethyl centralite system is worth while.

The foregoing suggestions are not intended to cover other than the immediate problems of interest in the study of dry-processed double-base colloidal propellants. There is a large field of research to be investigated in the development of new types of smokeless propellants that show relatively small influences of pressure, temperature, and transfer of radiant energy upon burning rate. Furthermore, there is the field of liquid propellants which, in the opinion of the writer, will probably supplant solid propellants in nearly all large rocket-propelled devices. The caliber of the rocket for which transfer from solid to liquid fuels will prove advantageous has yet to be established, but it is probable that there will be a range of sizes in which the application will determine the choice of a solid- or a liquid-fueled device. It is hoped that an effort will be made to standardize simple artillery rocket motors in order that a relatively wide variety of heads and stabilizing equipment can be used with a given motor.

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PART III

ROCKET ORDNANCE: THERMODYNAMICS AND RELATED PROBLEMS

By *R. E. Gibson* [a]

PART III OF THIS VOLUME will be concerned principally with problems arising in the development of colloidal solid rocket propellants and is really a summary of many of the final reports issued from Allegany Ballistics Laboratory [ABL], which was operated by George Washington University under contract[b] with the Office of Scientific Research and Development with technical supervision by Section H, Division 3, NDRC. Much of the pioneering work was done by the Section H group working at the Naval Powder Factory, Indian Head, Maryland, from 1941 through 1943. In this phase of the work close cooperation was established with the Hercules Powder Company, which, under contract first with OSRD and later with the Ordnance Department, contributed greatly to the phases of the program lying between development and production. Laboratory experimental work and theoretical studies on propellants were carried on by groups at the Bell Telephone Laboratories, University of Minnesota, University of Wisconsin, and Duke University, which worked very closely with the central Section H Laboratory, first at Indian Head, afterwards at Allegany. All these agencies contributed to the developments described in the following. Notable contributions to the general subject were made by Section L, Division 3, NDRC, Division 8, NDRC, Division 1, NDRC, the Bureau of Ordnance, U. S. Navy, and the Rocket Development Division, Ordnance Department, U. S. Army. These are discussed systematically elsewhere and will only be referred to casually in this report.

The problems in the physical chemistry of rocket propellants discussed in this report all arose from very practical questions which had to be solved in the development of rockets. These problems fall into categories well known in physical chemistry, namely, thermodynamic problems, kinetic problems, and structural problems. In Chapters 9, 10, and 11 the studies of rocket propellants will be summarized under each of these headings, respectively, and in each chapter an attempt will be made to indicate, first, the practical problems in the functioning of rockets that were encountered, second, the problems in the physical chemistry of the propellants that arose from these functional problems, and, third, a summary of the results obtained. Chapter 12 will give a short summary of the application of these problems to internal ballistics. Chapter 13 will give a summary of the rocket propellants which were developed by V–J Day and will indicate lines along which progress will probably be made in the future. As far as possible, reference to the original detailed reports will be given.

The reader who is unfamiliar with rocket problems is urged to consult *Rocket Fundamentals*,[1] a composite report to which a number of development agencies contributed and in which a fairly complete but elementary exposition of the principles of rocket design and action is given.

Since this volume will be printed long after V–J Day, it is fitting to point out that a great deal of the work described here has been continued with excellent results since the NDRC activities stopped. Allegany Ballistics Laboratory has continued operations with a new contractor, the Hercules Powder Company, under contract with the Bureau of Ordnance, U. S. Navy. The results and techniques developed at the laboratory under NDRC have been applied and extended to the development of large rockets with solventless-extruded and cast double-base powder charges, and devices of great interest in the guided missiles program are being perfected. Reports from this laboratory should be consulted for the sequel to this summary. Furthermore, the laboratory and theoretical studies of propellants conducted at the University of Minnesota have continued under the auspices of the U. S. Navy. Reports from this university should also be consulted for the continuation of the work started by NDRC.

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[a] Director of Research, Allegany Ballistics Laboratory.
[b] Contract OEMsr-273.

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Chapter 8

TYPES OF ROCKET PROPELLANTS

By *R. E. Gibson*

8.1 JET PROPULSION, ROCKETS, AND PROPELLANTS

In modern civilian or military engineering there is a wide variety of devices for propelling projectiles or other vehicles. Although they may differ widely in their construction and in other superficial respects, practically all propulsion mechanisms have fundamentally the same basis: they depend on the conversion of the energy of a controllable chemical reaction into elastic energy of a gas, which is then converted, by a suitable mechanical device, into kinetic energy of motion in a given direction. The mechanical devices by which the elastic energy of the gases is converted into useful work, i.e., into the kinetic energy of the vehicles, vary in complexity from the locomotive or airplane engine to the simple gun barrel or rocket jet. The choice of engine depends mostly on the ultimate application; the rate at which energy must be supplied, the mobility of the apparatus, the number of hours of working life required, and other performance requirements must be balanced against economic factors in choosing the chemical reactants and the mechanical apparatus to be used. Few people would use nitrocellulose powder to fire a locomotive, and few would use coal to propel a large military missile.

8.1.1 Rockets

Probably the simplest device for converting the elastic energy of a gas into the directed kinetic energy of a vehicle is the jet engine. Like other motors, these jet engines depend for their energy on a chemical reaction which we may consider to be an oxidation reaction involving a fuel (the substance to be oxidized) and an oxidizing agent.

A rocket is a jet-propelled vehicle which carries with it all the components needed for the energy producing chemical reactions, i.e., both the fuel and the oxidizer. This characteristic differentiates the rocket from other jet engines such as the ram jet, the pulse (or reso) jet or the turbo jet, all of which draw their oxidizer from the atmosphere through which they pass. The ram jet and pulse jet draw in air simply by making use of the dynamic pressure produced by their motion through the air, whereas the turbo jet makes use of compressors driven by part of the energy generated by the motor.

8.2 ROCKET PROPELLANTS

The term *rocket propellant* is applied to the chemical substance or substances which react to produce the hot gases whose elastic energy is to be converted into the kinetic energy of motion of the projectile. There are two main types of rocket propellants: liquid propellants and solid propellants.

8.2.1 Liquid Propellants

Liquid propellants in turn fall into two main classes: bi-fluid systems and mono-fluid systems. In bi-fluid systems, which have found most common use to date, the oxidizer and the fuel are kept in separate tanks in the rocket and fed in proper proportions into a combustion chamber where they react. Such systems are relatively safe as regards hazards during storage or transit and permit a wide range of control of rate of gas evolution and temperature, because it is possible to control independently the supply of fuel and oxidizer. Typical oxidizers are nitric acid, hydrogen peroxide, and liquid oxygen, and typical fuels are aniline (or mixed aromatic amines), hydrazine, methyl alcohol, and gasoline. Mono-fluid rocket propellants are liquids which contain in themselves sufficient oxygen to give fairly complete oxidation of the other elements with evolution of heat, when a reaction is started. Although all such substances are of necessity thermodynamically unstable, a number of suitable propellants, such as nitro-methane and hydrogen peroxide, have been found which decompose at a negligible rate at ordinary temperatures and can, therefore, be handled with comparative safety. Nevertheless, precautions required to store and handle such propellants tend to offset the obvious engineering advantages to be gained when two liquids are replaced by one.

8.2.2 Solid Propellants

In *solid propellants* the fuel and the oxidizer are intimately mixed and in a condition to react rapidly, but controllably, when the necessary activation energy is supplied, usually by a device called an igniter. It is a necessary characteristic of all solid rocket propellants that the reaction (which is usually called the "burning") take place only on the exposed surfaces of the solids and that burning proceed in directions normal to the surfaces at a rate which is the same at all points.

For very large and long-range rockets such as the V–2, or for applications where good thrust control is required, as in a jet plane, liquid propellants possess overwhelming advantages over solid propellants. It must be noted, however, that the valves and plumbing systems in these large rockets are complicated and costly; in the V–2 rocket the fuel system must be capable of supplying about 270 lb of fuel and oxidizer per second. In smaller rockets, therefore, particularly where ease of handling and simplicity of design are important, solid rocket propellants have a field of application in which they are unrivaled.

During World War II the activities of Section H, Division 3, NDRC, were confined to rockets or jet-propelled devices weighing less than 200 lb. Its attention was, therefore, concentrated on solid propellants, and Part III of this report will be concerned only with this type of propellant.

8.2.3 Composite and Colloidal Propellants

Two main classes of solid propellants are recognized. In one class the oxidizer and the fuel are present as separate molecules, or as small crystalline aggregates intimately mixed and held together by adhesives designed to give suitable mechanical properties to the mass as a whole. These are called *composite propellants*, and the classical example is ordinary black powder where the oxidizer is potassium nitrate and the fuel is charcoal. During World War II considerable effort was expended in the development of new and improved composite propellants. Section H, Division 3, took no part in the actual development of these propellants but was active in testing them ballistically. The research and development work was done by Division 8, NDRC, and by the Guggenheim Aeronautical Laboratory, California Institute of Technology [GALCIT]. GALCIT developments were later extended and applied by the Aerojet Engineering Corporation. Three significant varieties of composite propellants were developed by these agencies. Division 8 produced composite propellants by the molding, solvent extrusion, and casting methods. GALCIT produced a number of cast perchlorate propellants. The preparation and properties of these propellants are given in Chapter 13. In smaller artillery rockets, composite propellants found relatively limited application, although the solvent-extruded composites gave the answer to a very urgent need that arose in connection with the infantry bazooka rocket.[1] On the other hand, the composite propellants, because of simplicity of manufacture and their desirable burning properties, proved to be extremely well suited to use in rocket motors where long burning times and large amounts of propellant were required. Indeed their only disadvantage arose from the smoke they produced.

The second class of propellants, and the class which found most extensive use in the artillery rockets of all nations engaged in World War II, comprises the *colloidal propellants* which have been used for years. In colloidal propellants, the oxidizer and the fuel are on the same molecule, and the solid itself is macroscopically homogeneous. Colloidal propellants consist essentially of a high polymer which is rich in oxygen and can undergo an exothermic reaction in which its elements are raised to a higher state of oxidation. The high polymer may be plasticized with oxygen-rich plasticizers which are metastable chemically, or with plasticizers which are essentially fuels. The plastic formed by the interaction of high polymer and the plasticizers gives a homogeneous mass in which suitable physical properties may be developed. Since the main stimulus for improving solid propellants for rockets came from the desire to throw heavier payloads faster and farther for military purposes, it is not at all surprising that rocket development agencies in all countries should have turned to con-

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ventional gun propellants for the first source of high-energy fuels. Of the various gun propellants available, the class called *double-base powders* proved most suitable, chiefly because of their ability to react reliably at relatively low pressures, 300 to 1,500 psi, and because it was found possible to fabricate them into "grains" of suitable shapes and sizes for rocket work. Single-base powders possess neither of these properties.

Double-base powders receive their name from the fact that they contain two explosive ingredients—one being a high polymer (up to now always nitrocellulose) and the other being a plasticizer, usually nitroglycerin; other explosive plasticizers have also been used, e.g., diethylene glycol dinitrate, DINA, and TNT. Generally speaking, the nitroglycerin forms between 30 and 45 per cent of the whole mass, the rest being nitrocellulose with varying amounts of auxiliary plasticizers such as ethyl or methyl centralite, triacetin, and dinitrotoluene, stabilizers such as ethyl centralite or diphenylamine, and inorganic salts such as potassium nitrate or potassium sulphate. In some very desirable double-base rocket propellants developed during World War II, the amounts of auxiliary plasticizers such as triacetin or centralite rose in amount to something between 5 and 20 per cent of the whole composition. In double-base powders the nitrocellulose is gelatinized with or without the help of an active volatile solvent by mechanical working. The resulting mass is a hard, hornlike, homogeneous, rigid colloid which obeys ideally the law of burning in parallel layers.

For rocket applications where short burning times and high accelerations are required, double-base powder gelatinized with the help of an active volatile solvent is very suitable because of the high physical strength that may be developed in the grains. The "solvent process," although also being advantageous because of the ease and relative safety in manufacture, is severely limited in application, since the removal of the solvent sets an upper limit to the "web" thickness (minimum dimension of grain) that may be obtained. In the "solventless process" the double-base powder is gelatinized by severe working on heated rolls without the aid of an active volatile solvent. This method is particularly advantageous when longer burning times are required. In the solventless process the colloided powder is formed into grains by extrusion under high pressure at elevated temperatures, and essentially the only upper limit to the web thickness that can be made available is that imposed by the size of press that is safe and practical to operate. The process is not suitable for making single-base powder but is well suited to the manufacture of double-base powder containing less than 60 per cent nitrocellulose.

The solventless process was developed in Germany prior to World War I and was introduced into Great Britain and France shortly thereafter. It was extensively used in Russia at least as early as 1931. Prior to World War II only a small amount of solventless double-base powder was used in the United States, and this only in sheet form for use in trench mortars. No apparatus existed for extruding solventless powder into cylindrical grains, and indeed the industry exhibited a strong prejudice against setting up such an operation. Thus, while the rocket developers in Great Britain found in 1935 a ready production source of a high-power solid rocket propellant in the factories used for making solventless cordite for the Royal Navy, the American rocket developer found himself starting from scratch, or rather several yards behind the line. It is not too much to say that the setting up of a solventless powder industry came directly as a consequence of the visits of NDRC investigators to Great Britain.

This introductory chapter concludes with a chart illustrating the various levels of problems connected with the development of a complete artillery rocket. It is designed to give the reader a general idea of the problems encountered and the equipment and facilities needed for their solution.

TABLE 1. Research and facilities required in the development of a rocket motor.

ROCKET—COMPLETE ROUND

Problems	Facilities
Tests of safety & reliability in flight Assessment of performance, velocity, acceleration, stability, accuracy, & range The flight of rockets	Flight ranges Barricaded launchers Chronographs Ballistic cameras Surveying & spotting apparatus Theoretical group for analyzing data & solving mathematical problems

Metal Components	*Propellant System*
Payload—shell, explosive, & fuze Motor—powder chamber, venturi, & traps for supporting powder charge Fins for stabilizing flight	Powder charge Ignition system

Development of Metal Parts

Problems	Facilities
Design of head & fuze for optimum effect Design of chamber with minimum weight to withstand high pressures & temperatures of powder gases Design of complete round to give ease of manufacture & simplicity of loading & assembling Design of complete round including fins to give low drag, high stability, and minimum dispersion	Engineering design & drafting group Metal working shops Mechanical testing laboratory Procurement of metal parts Inspection

Development of Propellant Charge

Problems	Facilities
Choice of propellant Design of sizes & shape of propellant Methods of support of propellant Control of pressure & thrust over the complete temperature range Control of burning time Effect of climatic storage on the propellant system Control of gas flow in rocket Design of ignition system	Powder manufacture, charge preparation, & loading Static firing ranges with facilities for temperature control & instruments for measuring pressure & time, thrust & time, temperature & time Temperature cycling houses Control chemical laboratory Testing under conditions of rough usage Facilities for accumulation & application of knowledge of internal ballistics

Research on Metal Parts

Problems	Facilities
New materials for rockets Application of new engineering techniques to rocket manufacture Protection of metal parts, e.g., thermal insulation Control of nozzle erosion	Engineers in contact with engineering companies to develop new materials Tests of products under firing conditions Facilities for heat treating steel Facilities for handling materials other than steel

Research on Propellants

Problems	Facilities
Search for new propellants of suitable burning & mechanical properties Investigation of new charge shapes & means of preparation Study of the mechanism of burning of rocket propellants & the effect of environment on the burning Study of the flow & thermodynamic properties of propellant gases Study of new methods of fabrication of propellant charges	Source of powders of controlled compositions Static firing ranges for studying burning properties under different conditions of pressure, temperature, gas flow, & environment Lab for study of the chemical, physical, & mechanical properties of propellants Facilities for preparing propellant charges in a wide variety of shapes & sizes Theoretical group for developing & understanding internal ballistics of rockets

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Chapter 9

THERMODYNAMIC PROBLEMS

By *F. T. McClure* [a]

9.1 ROCKET ACTION—THRUST—SPECIFIC IMPULSE OR EFFECTIVE GAS VELOCITY

Sufficiently fine analysis of any propulsion system will resolve it into an example of Newton's third law of motion, namely, "To any action there is an equal and opposite reaction." Rocket propulsion is a particularly simple and direct practical example of this law. The rocket chamber or motor exerts a force on the gases contained therein, causing them to be expelled to the rear. This, if one wishes, is the action. In turn, the gases exert an equal force (in the opposite direction) on the rocket, causing it to be propelled forward. This, then, is the reaction.

One may guess (and in fact it is a consequence of Newton's second law of motion) that, for rocket motors of the same configuration operating under the same pressure conditions and using the same fuel, the thrust (F) will be proportional to the mass rate of exhaust of fuel. The proportionality constant is generally called the *specific impulse* (I) or the *effective gas velocity* (V_E) depending on the units in which it is expressed, so that

$$F = \dot{m}I$$

or

$$F = \dot{m}V_E \qquad (1)$$

where $\dot{m}$ is the mass rate of discharge of fuel. In this country, I is usually expressed as the pounds force for each pound per second mass rate of discharge, while V_E is expressed in feet per second. Then

$$V_E = 32.16I.$$

Because of the discharge of the fuel, the mass of a rocket decreases during the acceleration or burning time. If W is the mass of the rocket then $dW/dt = -\dot{m}$, and according to Newton's second law of motion

$$F = \dot{m}V_E = -V_E\frac{dW}{dt} = W\frac{dV}{dt}, \qquad (2)$$

where V is the velocity of the rocket. Integration of the last of equations (2) from the point of initiation of the thrust to the point at which the fuel is consumed leads to a final velocity [b] given by

$$V_0 = V_E \ln\left(1 + \frac{m_0}{M}\right), \qquad (3)$$

where m_0 is the original mass of fuel and M is the mass of the rocket without fuel.

Equation (3) clearly exposes the significance of the effective gas velocity or specific impulse to rocketry. Obviously, fuels capable of producing high specific impulse are most desirable, particularly for very high-velocity or long-range rockets. As discussed in the next section, the specific impulse of a fuel is determined partly by the operating conditions (pressure), partly by the motor geometry (expansion ratio), and largely by the thermodynamic properties (heat capacities, molecular weight, and temperature) of the propellant gas which it generates. It is through the specific impulse, then, that the thermodynamic properties of a fuel provide a measure of its potential.

It should be emphasized that the thermodynamic properties of a fuel are not the only properties of significance in determining its desirability. This may be seen by further examination of equation (3). A fuel must be packaged, and the weight of the package or container is included in M. The greater the weight of container necessary for a given weight of fuel, the more difficult it is to achieve a high ratio, m_0/M, of fuel weight to empty rocket weight. The container weight, however, is largely determined by the volume, and thus a high-density fuel has the advantage of a lower ratio of container weight to fuel weight. For example, the high specific impulse of the liquid hydrogen-liquid oxygen combination (due to the low molecular weight of the gases gener-

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[b] These equations neglect the effect of gravity and air resistance, both of which must be considered in dealing with high-velocity, long-range rockets.

ated) is partially nullified by the large tanks required for the hydrogen (because of its very low density). Such considerations apply to both solid and liquid fuels.

Other properties of fuels are also of importance in determining their usefulness. In particular, the ease, rapidity, and uniformity with which the complete conversion of the fuel into the propellant gas can be accomplished is important in determining the weight of the combustion chamber, which is also part of M in equation (3). This is again true of both liquid and solid fuel but is more strongly felt in the latter case because here the combustion chamber is also the container for the fuel. Such problems fall in the field of "interior ballistics."

It may also be worth noting that rockets use large quantities of fuel so that the ease, cost, and hazard associated with the manufacture, storage, transportation, and handling are important considerations in choosing a fuel.

9.2 THE CALCULATION OF THE SPECIFIC IMPULSE—THE REDUCED SPECIFIC IMPULSE

More careful study of the flow of gas from a rocket motor not only verifies the assumptions of the preceding section but also elucidates the dependence of the specific impulse on the thermodynamic properties of the propellant gas, the nozzle geometry, and the operating conditions. Such a detailed analysis is carried out in Chapter II and Appendices 2 through 8 of reference 1. An important result is that the specific impulse of a fuel-motor combination can be separated into a product of $\sqrt{nRT_c}$ and a function of γ, P_a/P_c, and A_e/A_t. Here, n is the inverse of the molecular weight, T_c is the absolute temperature, γ is the ratio of the heat capacities at constant pressure and constant volume, and P_c is the pressure of the gases in the combustion chamber, whereas P_a is the pressure of the surrounding atmosphere, A_e is the area of the nozzle exit, A_t is the area of the nozzle throat (narrowest section), and R is the universal gas constant. Because of this separability the quantity

$$\frac{I}{\sqrt{nRT_c}},$$

which is called the *reduced specific impulse*, is independent of n and T_c, and therefore may be tabulated or graphed as a function of the pressure ratio (P_a/P_c), the expansion ratio (A_e/A_t), and γ without reference to the molecular weight or temperature of the gas. In reference 2 the reduced specific impulse is tabulated and graphed over a wide range of values as a function of the pressure ratio and expansion ratio for each of the values 1.15, 1.20, 1.25, 1.30, 1.35, and 1.40 for γ. The graphs for $\gamma = 1.20$ are reproduced in Figure 1 as a sample. This report [2] also includes sample calculations and a summary of formulas with provisions already made for appropriate units, so that it becomes a simple matter to estimate the specific impulse for a given pressure ratio and expansion ratio providing n, T_c, and γ for the fuel are known.

A point of caution must be emphasized here. As Figure 1 indicates, the specific impulse increases with increasing expansion ratio until it reaches a maximum. This maximum occurs at the point where the exit pressure is just equal to the pressure of the surrounding atmosphere. It must not be concluded, however, that a large expansion ratio can be obtained by "opening up" the nozzle rather than by increasing its length, thus avoiding a penalty in nozzle weight. The graphs given neglect the sideways motion of the gas in the nozzle, which contributes nothing to the thrust. This effect is only negligible providing the divergence of nozzle is not too great.

9.3 THE DISCHARGE COEFFICIENT

As shown in reference 1, the mass rate of discharge of gas from the rocket motor may be expressed in the form

$$\dot{m} = C_D A_t P_c, \tag{4}$$

where C_D, the discharge coefficient, is given by

$$C_D = \left(\frac{2}{\gamma + 1}\right)^{(\gamma+1)/[2(\gamma-1)]} \frac{\sqrt{\gamma}}{\sqrt{nRT_c}} \tag{5}$$

and thus is determined by the thermodynamic properties of the gas in the rocket chamber. Sample calculations in appropriate units are given in references 1 and 2.

According to equation (5) the discharge coefficient is independent of geometry of the rocket motor. Actually there is a slight dependence on the geometry through the ratio of throat area to the free area of chamber (i.e., the cross-sectional area not

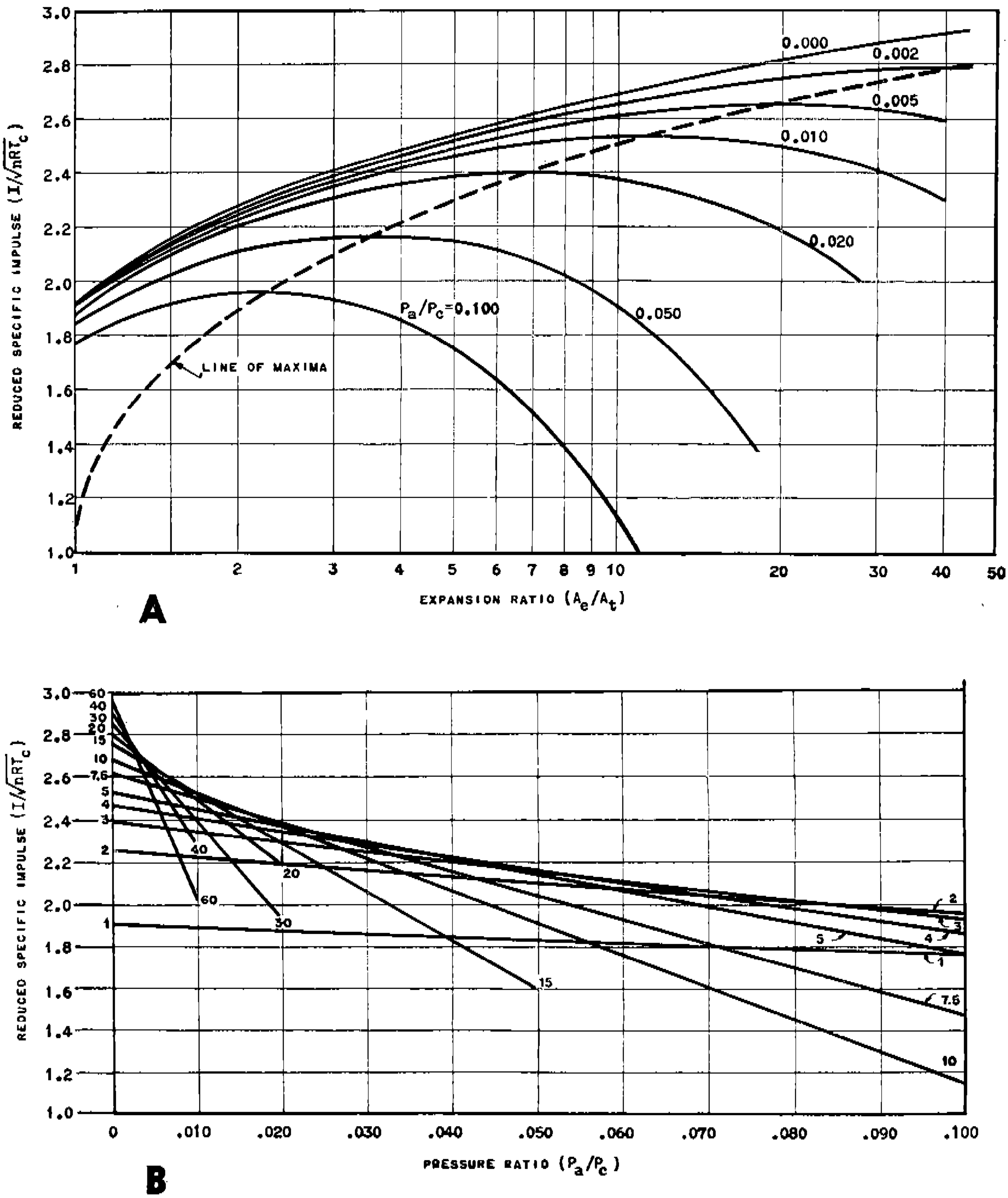

FIGURE 1. Properties of a propellant gas with $\gamma = 1.20$. A. Reduced specific impulse vs expansion ratio at various pressure ratios. B. Reduced specific impulse vs pressure ratio for various expansion ratios.

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occupied by propellant, etc.,—referred to as the "port area"). This dependence is due to the pressure drop and velocity gradient in the combustion chamber. The effect is discussed in some detail in Appendix 6 of reference 1.

9.4 THE THRUST COEFFICIENT

Frequently it is advantageous to express the thrust in terms of the chamber pressure according to the equation

$$F = C_F A_t P_c, \tag{6}$$

where C_F is known as the thrust coefficient. By use of equations (1) and (4) one may obtain

$$F = C_D I A_t P_c$$

so that

$$C_F = C_D I. \tag{7}$$

Further, combining equations (5) and (7), one obtains

$$C_F = \left(\frac{2}{\gamma + 1}\right)^{(\gamma + 1)/[2(\gamma - 1)]} \sqrt{\gamma} \left(\frac{I}{\sqrt{nRT_c}}\right). \tag{8}$$

It will be noticed that the last factor on the right of equation (8) is just the reduced specific impulse, which is a function of γ, P_a/P_c, and A_e/A_t. Thus the thrust coefficient is a function of γ, P_a/P_c, and A_e/A_t and is independent of the molecular weight and temperature of the propellant gas.

The graphs of reduced specific impulse in reference 2 may be used to compute the thrust coefficient through equation (8). A sample calculation is given in the reference.

9.5 CALCULATION OF THE THERMODYNAMIC PROPERTIES OF THE GAS FROM THE COMPOSITION OF THE FUEL

From the preceding sections it is clear that the important properties of the propellant gas, from the standpoint of specific impulse, discharge coefficient, etc., are the values of γ, n, and T_c. Ideally, the temperature of gases in the combustion chamber is the so-called isobaric adiabatic flame temperature, which is related in a simple manner to the higher isochoric adiabatic flame temperature characteristic of the reaction in a closed vessel. Both adiabatic flame temperatures are, in theory, calculable from the thermodynamic properties of the fuel. Their definitions and relationship are discussed in Appendix 2 of reference 1.

In principle, the computation of the thermodynamic properties from the composition of the propellant is a straightforward problem in classical thermodynamics. In practice, however, it is the developments of the last twenty years which have made the solution of the problem possible. The development of quantum statistical mechanics and the analysis of band spectra has provided the only satisfactory method now available for estimating the heat capacities of the constituent gases at the temperatures as high as those encountered in guns and rockets (of the order of 2500 to 4000 K). Further, these developments, supplemented by data obtained from modern low-temperature calorimetry, have made possible the calculation of equilibrium constants under conditions such that accurate direct measurements are experimentally impractical.

It would be far beyond the scope of this report to attempt to outline, in any detail, the process of calculating the thermodynamic properties of a propellant gas. Such an outline represents a sizable manuscript in itself. A schematic block diagram is, however, given in Figure 2 and serves to indicate the general steps in the process. Rather detailed discussion of the methods of building up the requisite thermodynamic tables is given in reference 3. The reference also provides such tables and carries through in detail several examples of the application to specific propellants. References 4 and 5 apply these methods to detailed calculations for a number of other propellant compositions.

Actually, references 3, 4, and 5 are concerned with finding the isochoric flame temperature (of interest in gunnery) and the properties of the gas under these conditions. However, as indicated in Appendix 2 of reference 1, the conversion from the isochoric to the isobaric flame temperature is a relatively simple matter.[c]

Although the methods of reference 3 are capable of considerable accuracy, they are somewhat laborious, and for this reason simple, more approximate methods of estimating the thermodynamic proper-

[c] In references 3, 4, and 5 the conversion is particularly simple, since the isobaric flame temperatures are essentially the temperatures on the Mollier charts of enthalpy versus entropy at which the enthalpies are equal to the "enthalpy constants" (symbol H_1 in references 3 and 5 and symbol A in reference 4).

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ties of the propellant gases have been developed. These developments and illustrations of their use are described in references 6 and 7. Summaries and tables are available in Appendix 8 of reference 1 and in the Appendix of reference 2.

It must be remembered that the justification for the simple methods of reference 6 is based on agreement with the more complete methods of reference 3. In this sense the simple method may be considered an "interpolation system" for the more complete treatment. In the case of the application to propellants of composition widely different from those previously treated, it would appear to be wise to recheck the simple scheme against the complete one, modifying the constants of the former as necessary to bring it into agreement with the latter.

The success of the thermodynamic calculation discussed in this section is essentially limited to fuels composed almost entirely of compounds of carbons, hydrogen, oxygen, and nitrogen. The inadequacy with respect to fuels containing appreciable quantities of other kinds of constituents (such as composite propellants which contain large proportions of inorganic constituents) does not lie within the methods described but, rather, is due to the almost complete lack of adequate basic thermodynamic and spectral data for these other constituents and their reaction products. In this sense, thermodynamics is like a large production machine; poor raw material leads to a poor finished product, and absence of raw material leads to no product at all.

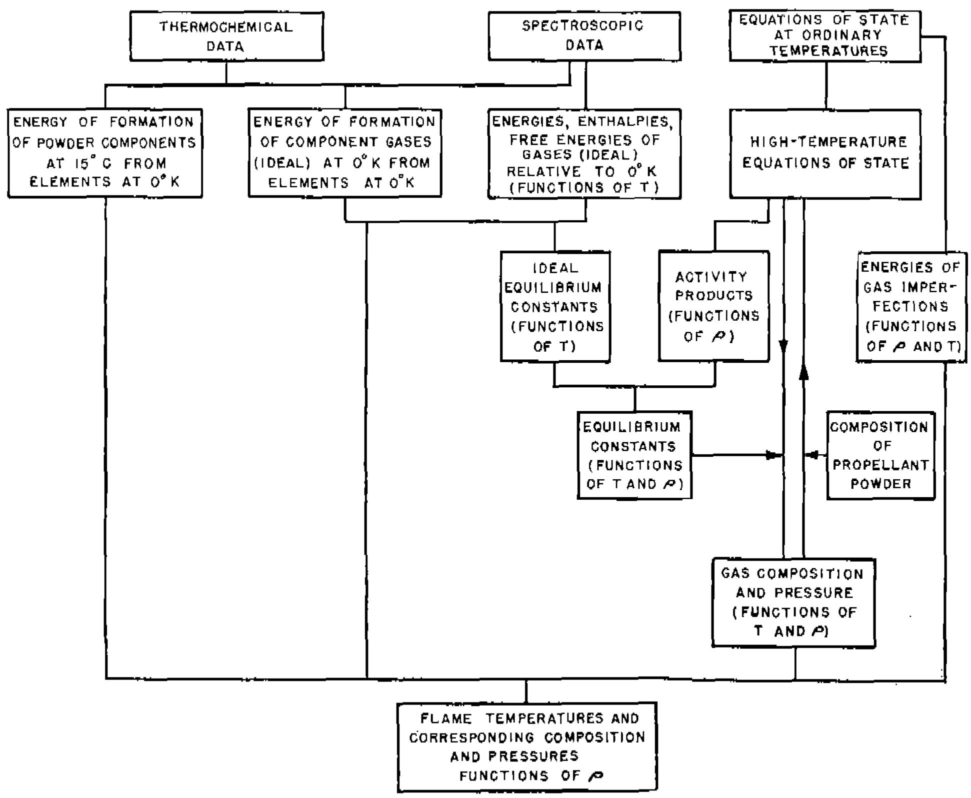

FIGURE 2. Block diagram of procedure for calculating the thermodynamic properties from the composition of the fuel.

9.6 HEAT LOSS, INCOMPLETE REACTION, POWDER LOSS, AND OTHER MODIFYING FACTORS

The preceding sections deal with the theoretically ideal performance of a rocket motor. In practice, many factors arise which prevent the attainment of such ideal performance. Detailed description of these factors and their effects would require a lengthier dissertation than can be given in this report; however, since their recognition and minimization represent a large part of the science of rocketry, a brief summary is presented in Table 1. Most of the information in this table may be in-

[redacted]

ferred from the definitions of the quantities involved, although in some cases other sources must be called upon.

Some brief comments and qualifying remarks may be useful in understanding Table 1. The effect of heat loss on the specific impulse and discharge coefficient arises largely from the lowering of the flame temperature, although there is a slight effect due to the accompanying small increase in γ. The thrust coefficient, being affected essentially solely through the change in γ, is decreased much less markedly than the impulse. It may be noted that at higher expansion ratios the influence of changes in γ is greater, so that at very high expansion ratios (such as might prove useful in very high-altitude propulsion) the change in thrust coefficient may become somewhat more marked than in the case of typical artillery rockets of World War II.

The effect of incomplete reaction is similar to heat loss except that increasing incompleteness of reaction does not necessarily mean uniformly increasing γ, so that the thrust coefficient may not decrease uniformly. However, when the incompleteness of reaction is not extreme, it appears likely that the deviation of the thrust coefficient from the theoretical will be negative and relatively small. Increased operating pressures will increase the completeness of reaction simply because the gas phase reaction proceeds more rapidly at higher pressures. Incomplete reaction is generally more predominant with "cool" (low flame temperature) than "hot" powders.

TABLE 1. Deviations of static* measurements from theoretical values.

	Deviation from ideal			Influence of operating conditions on deviation		
Modifying factor	Specific impulse	Discharge coefficient	Thrust† coefficient	Operating pressure	Initial†† powder temperature	Comments
Heat loss	–	+	Small –	Generally not large	Generally not large	Uniform from shot to shot
Incomplete reaction	–	+	Small, probably –	Effect decreases with increasing pressure	Generally negligible	Uniform from shot to shot
Powder loss	–	+	Negligible	Increases with increasing pressure	Generally marked at very high and very low temperatures	Erratic from shot to shot
Poor nozzle approach	–	–	–	Small	None	Nozzle approach generally badly eroded during shot
Poor nozzle expansion section	–	None	–	Depends on design	None	Excessive divergence of cone—roughness or poor contour leading to non-adiabatic flow
Pressure gauge recording high	None	–	–	None	None	Emphasizes the importance of frequent gauge recalibrations
Thrust gauge recording high	+	None	+	None	None	
Poorly controlled instrumentation	Variable	Variable	Variable	None	None	General erratic behavior with little or no correlation with operating conditions

* That is, with rocket motor held in test stand.
† See qualifying remarks in context.
†† Operating pressures are generally increased by increasing initial powder temperature. Care must be taken in separating pressure and temperature effects.

Powder loss in a given rocket motor increases with increasing pressure because of the greater stresses thus applied to the charge. Superimposed on this, however, there is often a marked increase in loss at high powder temperatures (where the pressure is generally high) due to the "softening" of the grains, and at low powder temperatures (where the pressure is generally low) due to increased "brittleness" of the grains, which results in tendency to fracture under the shock from the igniter. Powder loss does not influence the thrust coefficient unless there is significant temporary blocking of the nozzle, in which case the result is more apt to be a motor

[redacted]

rupture than a recorded deviation of the thrust coefficient.

Excessive roughness in the nozzle approach may decrease the efficiency through skin friction. Sharp edges may produce excessive turbulence or a "vena contracta" which reduces the effective throat area, with the result that best advantage of the expanding cone is not obtained.

Roughness or poor contours in the expanding section of the nozzle may lead to the development of "shock waves" in the nozzle with lowering of the specific impulse. Such difficulties tend to become more predominant at higher expansion ratios.

For the general run of artillery rockets of World War II the observed specific impulses ran from about 5 to about 10 per cent below the theoretical (except in cases of large powder losses). The deviations appeared to be largely due to heat loss, although imperfect nozzle design probably made some contribution. Thrust coefficients about 2 to about 5 per cent low appeared to be the general observation. The deviation is again probably attributable to heat loss and imperfect nozzle design.

9.7 THE ATTAINABILITY OF HIGH SPECIFIC IMPULSE FUELS

It will be noted from the discussion of Section 9.2 that the principal properties of a fuel which determine its specific impulse are the molecular weight and temperature of the propellant gas, both of which enter the specific impulse as their square roots. A typical rocket fuel of World War II might have, for example, a flame temperature of 3000 K, an average molecular weight of 25, with a specific impulse of, say, 210 under ordinary operating conditions.

Consider the possibility of a fuel better by a factor of 3 than such a conventional fuel. Suppose the improvement were to be obtained by an increase in temperature. Then a temperature of *27000 K* would be required. Aside from the difficulties of finding a chemical reaction to produce such a temperature, one can imagine the problem of finding materials from which to form rocket walls and nozzles, capable of withstanding such conditions.

On the other hand, let the improvement be sought in the form of a reduced molecular weight. Hydrogen, with a molecular weight of 2, is the lightest gas available to us. On this basis we might expect to obtain an improvement by a factor of about 3.5. Actually the improvement would be somewhat less than this because of the weight of an appropriate heater for the hydrogen and the low density of liquid hydrogen (see Section 9.1).

It is apparent, therefore, that a fuel improved by a factor of, say, 3 over conventional fuels (which is, of course, a sizable improvement) will represent an outstanding achievement, whereas improvements much greater than this would appear to require revolutionary developments in the science of reaction propulsion.

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Chapter 10

KINETIC PROBLEMS

By *R. E. Gibson*

10.1 INTRODUCTION

Two very important questions in the design and functioning of rockets focus our attention on the chemical kinetics of the burning of the propellant. Both these questions are connected with the equilibrium pressure established in the rocket chamber. The first question, one of engineering design, arises from the necessity of making rocket chambers as light as possible, since they really amount to dead load, and any reduction in weight of the dead load means a gain in payload or in velocity. This puts up to the designer of a rocket the question of how to make his rocket motor as light as possible and at the same time strong enough to withstand any internal pressure likely to be developed. Control of the internal pressure is very important, therefore, from the viewpoints of efficiency and safety of design. The second question arises from the fact that the thrust of a rocket, and hence the acceleration it receives, is given by the product of the area of the thrust, the throat coefficient, and the internal pressure, $F = A_t C_n P$. Since A_t and C_n are substantially constant, we see that the internal pressure determines the acceleration of the rocket and hence its trajectory and external ballistics, particularly the value of the gravity drop during acceleration. The second question is, therefore, can the internal pressure be controlled within tolerance compatible with required ballistic performance.

The equilibrium pressure in a rocket chamber is determined by a balance between the rate at which gas is produced by the propellant and the rate it is exhausted through the nozzle. The rate at which the propellant produces gas is proportional to the area of burning surface and the *linear rate* at which the burning surface progresses. The effect of composition, pressure, temperature, and other environmental factors on linear rates of burning, therefore, takes on a very practical significance. It should be mentioned in passing that the term "burning" used in this connection should not be confused with burning in the sense commonly used, namely, to denote interaction of the substance being burned with atmospheric oxygen. In the "burning" of solid propellants, as it takes place in rockets, atmospheric oxygen plays no part, although it has been found that the accidental presence of atmospheric oxygen may lead to confusing results in experimental studies.[1a] The term "burning" when applied to a propellant refers to the extremely complex chain of reactions which go on when the molecules in the system, for example, nitrocellulose-nitroglycerin stabilizers, undergo rearrangements to give oxides of carbon, water, nitrogen, and small amounts of other simple molecular species.

10.2 LAW OF BURNING—EFFECT OF PRESSURE ON LINEAR RATES

A grain of propellant burns on all exposed surfaces, and the burning surface progresses into the body of the grain at a linear rate which is the same at all points provided that the powder is homogeneous and that external conditions are uniform. This law is often called the law of burning in parallel layers. The linear rate of burning does, however, depend on the pressure of the gas over the propellant, the original temperature of the grain, its chemical composition, and, to a lesser extent, on factors which will be discussed later.

Two equations have been used extensively for expressing the linear rate of burning of a propellant as a function of pressure:

$$r = a + bP, \tag{1}$$

$$r = cP^n, \tag{2}$$

where r is the linear rate of burning, P the pressure under which the powder burns, and a, b, c, and n are empirical constants.

Considerable thought has been given to the adequacy of one or the other of these equations to fit the experimental data. Results for some powders are better fitted by (1) than by (2), and for other powders the reverse is the case. Neither equation fits within experimental error over a very large

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range of pressure, but either usually gives an excellent fit over a range of several thousand pounds per square inch. This subject is discussed in several reports.[2–6] Several important powders developed during World War II exhibit a pressure dependence of the linear burning rate that is not well expressed by either (1) or (2).[3,9] If, however, we use either of these equations to derive a formula for the equilibrium pressure in a rocket motor, we arrive at an equation which is not misleading and does bring out the role of the various factors involved.

Equations in terms of both burning rate laws are derived in *Rocket Fundamentals*.[23a] Equation (3) gives the form corresponding to the burning rate law (2) and, being the simpler to follow, is quoted here.

$$P = \left[\frac{Sc(\rho - \rho_g)}{A_t C_D}\right]^{1/(1-n)}. \tag{3}$$

In equation (3), P is the equilibrium pressure, S is the area of the burning surface of the propellant, ρ is the density of the solid propellant, ρ_g is the density of the propellant gas in the chamber, A_t is the area of the throat of the rocket, C_D is the discharge coefficient of the gas, and c and n are the constants in the burning law equation. In Chapter 12 the effects of other factors influencing the steady-state pressure are discussed.

10.3 KINETIC FACTORS INFLUENCING THE EQUILIBRIUM PRESSURE

Equation (3) shows at once that a stable equilibrium pressure can be generated and maintained in a rocket only if n for the propellant is considerably less than unity. If n is equal to 1, $1/(1 - n)$ becomes infinitely large, and any small change in one of the factors within the bracket will cause an infinitely large change in pressure. A rocket could not be designed under such conditions. On the other hand, if n is zero, an ideal state is reached, because under such conditions the equilibrium pressure would vary only linearly with the quantities within the bracket. In general, if n is between zero and one, a stable equilibrium can be reached—the pressure will rise or fall to adjust itself to the equilibrium value. The values of n for the double-base powders available at the beginning of World War II lie between 0.7 and 0.8, close enough to unity to raise difficult problems in rocket design. If we assume an average value of 0.75 for the n of these powders we see that equation (3) becomes

$$P = \left[\frac{Sc(\rho - \rho_g)}{A_t C_D}\right]^4 \tag{4}$$

and that the internal pressure varies as the fourth power of the parameters within the bracket. The balance is a delicate one, for example, a rise of 10 per cent in the area of the burning surface will cause the pressure to rise more than 40 per cent. Changes in the other variables produce equally drastic effects.

We have discussed in Chapter 9 the limitations placed on A_t by port area and loading density considerations, and we have also shown that C_D depends on the thermodynamic properties of the propellant gas. It is unnecessary to discuss these quantities further here except to point out that, where n is large, the area of the throat of a rocket must be held within very close tolerances and that erosion during burning can easily upset the pressure balance in the rocket significantly. The quantities which concern us most in a consideration of the kinetics are n, S, and c.

10.4 THE PRESSURE EXPONENT

It will be seen at once that extremely practical considerations demand that a good propellant have a linear rate of burning which varies as little as possible with pressure, i.e., n should be as close to zero as possible in equation (2), or b and a should be as small as possible in equation (1). This requirement led at once to two lines of research: (1) an empirical study of the effect of composition changes on the pressure dependence of the rate of burning of a powder and (2) theoretical studies to develop an understanding of the burning process with a view to isolating the factors that determine n and finding out how to control them. The theoretical studies progressed to the point where a satisfactory general theory of the mechanism of burning double-base powder was formulated. The Universities of Minnesota and Wisconsin, Division 8, NDRC, and the British investigators made major contributions in this field (see bibliography listed in reference 4), but it cannot be said that any really useful means of reducing the pressure dependence of the rate of burning has yet come from these studies. The empirical studies which will be outlined later in this chapter were more successful, and by 1945 a number

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of double-base powders with very low pressure exponents over a given range of pressure were discovered. Of these, powders H–4 (T–2), L 4.8, and G117B were the most noteworthy examples.[3]

10.5 THE CONSTANT c

In addition to depending on the pressure, the burning rate of a grain of powder depends on its composition, its temperature, and the velocity of the gas stream in which it finds itself. The radiation falling on the powder also influences the burning rate, but, since this effect works by raising the powder temperature, it need hardly be considered to be an independent one.

When powders whose compositions differ widely are examined, we find that they give values of both c and n in equation (2) which are different. If, however, one examines a series of powders whose compositions do not differ widely—for example, manufacturing variations of the same basic formula —we find that n may be taken as the same for all the powders and the variations in burning rate may all be absorbed by variations in the constant c. Likewise, change of temperature has little effect on n but does change the constant c.

We may assume, therefore, that manufacturing fluctuations in composition and variation in ambient temperature affect the equilibrium pressure in a rocket by changing c in equation (2). If n is large, then changes in c will produce magnified changes in P. This, of course, again emphasizes the value of reducing n, but, if such a reduction is not possible, every effort should be made to reduce the variations of c as a result of composition and temperature fluctuations. These considerations lead again to the need of empirical and theoretical knowledge about the effect of composition and temperature on the burning rates of powders at a given pressure.

10.6 THE AREA OF THE BURNING SURFACE

If a constant chamber pressure throughout the entire burning time of the propellant is desired, and this is generally required for the most efficient design, it will be seen that the area of the burning surface of the powder must remain constant within very narrow limits. When the propellant obeys exactly the law of burning in parallel layers, it is a relatively simple matter to calculate the burning surface area at any instant if the original geometry of the grain is known, and it is possible to arrange this geometry in such a way that the burning surface does remain constant within the desired limits throughout the reaction. A singly perforated cylinder burning only on the external and internal cylindrical surfaces is a simple example of a grain whose burning area remains constant, i.e., a neutral-burning grain. The cylinder burning on the ends as well as the inner and outer surfaces would be a regressively burning grain since the area of the burning surface would decrease during the process. A number of sufficiently neutral grains were developed during World War II. It might be added that the increase in port area during burning causes the pressure curve to be regressive even for a grain having a constant burning surface. In cases when this effect is particularly large, it is advantageous to have the charge arranged so as to produce an increase in surface during burning to give a more constant pressure.

10.7 RATES OF BURNING OF DOUBLE-BASE POWDERS

The chief experimental work involved in the study of the kinetics of the burning of rocket propellants consisted of making reliable measurements of the linear rates of burning of powders of different but known compositions at different pressures and temperatures. Other experimental investigations concerning the effects of radiation[7,8] and of rate of gas flow[4] on the burning rates were also made. At the outset of the work the opinion was held by some workers with apparent justification that small-scale determinations of burning rates were of little value in the prediction of the ballistic behavior of a propellant in full-scale rockets. However, results of subsequent investigations showed that this opinion was not well founded and that small-scale experiments give useful information about propellants provided that proper account is taken of all the variables involved. An example of the practical application of small-scale experiments is to be found in the report on the development of a smokeless propellant for the JATO unit.[9] This subject is discussed further in Chapter 12.

Three distinct methods were used for determining the burning rates of powders: (1) closed bomb method, (2) vented vessel method, and (3) burning strand method. For the same powder, these three methods all gave values of the burning rate at a given pressure and temperature which were reconcilable, although the task of reconciling them was accomplished only after considerable study and consequent gain in knowledge of the processes involved. The methods will now be described.

10.7.1 Closed Bomb Method

This method has been extensively used in connection with gun propellants. A sample of powder of known geometry is enclosed in a heavy-walled steel vessel, or bomb, capable of withstanding upwards of 100,000 psi. The bomb is provided with a water jacket to control its temperature, and with a fast-responding pressure gauge by which the pressure is recorded as a function of time during the burning. It is now common practice to use a piezoelectric gauge with amplifier and oscilloscope and to record pressure and change of pressure with time simultaneously.

In an experiment the bomb is closed tightly to prevent gas leakage and the powder ignited. After proper corrections for cooling, the maximum pressure and the change of pressure with time give the rate of gas evolution, and this information combined with a knowledge of the geometry of the grain enables one to calculate the linear burning rate at any pressure in the region covered.

The possibilities of the closed bomb for studying rocket propellants were explored at Duke University and at ABL and several reports are available.[10] The method is valuable for giving rates of burning when the linear mass flow of gas over the propellant is essentially zero. In general, however, the closed bomb is less useful than the other methods, chiefly because its accuracy is best at high pressures and it is not well adapted to giving accurate results at low pressures—below 2,000 psi, the region of interest in rocket work.

One very interesting phenomenon was observed when singly perforated grains were burned in closed bombs, namely, that high-frequency vibrations were set up during the burning, especially on records plotting dP/dT. These vibrations were stopped if a steel rod similar to a trap wire in a rocket was slipped through the perforation.[10] The phenomenon is akin to the "resonance effect" found in rockets and mentioned in Chapter 12.

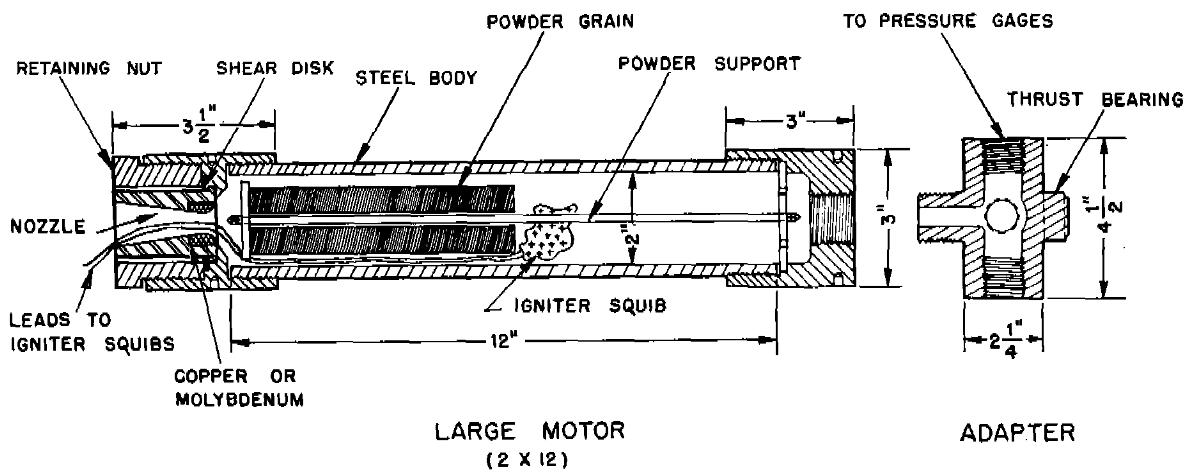


FIGURE 1. Small-scale experimental rocket.

10.7.2 Vented Vessel Method

In this method the grain of the propellant is burned in an experimental rocket motor fitted with a venturi to give the desired equilibrium pressure. For any series of experiments several motors and a large number of venturis of different sizes are required.[4,5,15] In order to cut down effects of high gas velocity on the burning rate, the motors should be so designed that the free port area is much greater than the area of the throat of the nozzle. Special

[redacted]

apparatus was used for extruding the powder into suitable grain sizes for this work, and the shapes and sizes were carefully controlled by machining and measurement. The temperature was controlled by conditioning the motor and propellant in a suitable thermostat before firing, and the pressure as a function of time was measured by rapidly responding Bourdon gauges [12] or strain electronic gauges.[13] Special precautions for getting rid of the exhaust gases and barricades to confine the results of explosions were needed. The pressure-time curves at different temperatures and the geometry of the propellant grain are the primary data and suffice to give the burning rate as a function of pressure and temperature. A typical experimental rocket motor, a pressure-time curve, and a graph showing burning rate as a function of pressure are shown in Figures 1, 2, and 3.

By means of this technique, several hundred powders covering a wide range of compositions of double-base and composite propellants were examined at Indian Head and Allegany Ballistics Laboratory. The results are to be found in references 3, 5, and 15. In some cases time permitted only the gathering of fragmentary data, and these results are to be found in the files of Allegany Ballistics Laboratory.

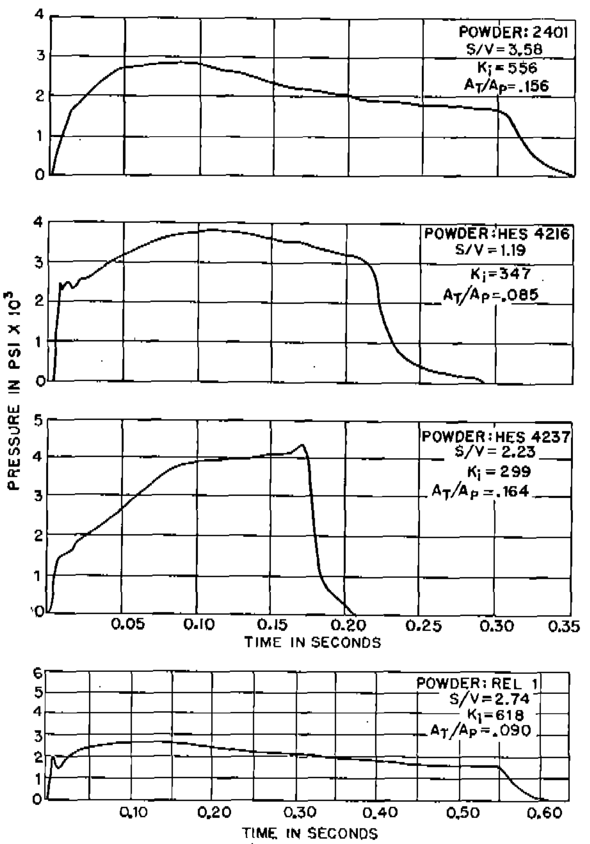
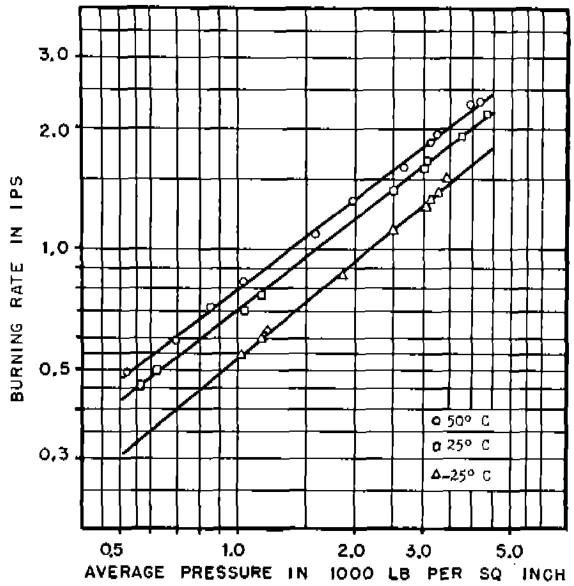


FIGURE 2. Typical pressure-time curves obtained from burning of powder in vented vessels.



POWDER COMPOSITION	A-68
NITROCELLULOSE	57.55
INCLUDING %N	13.21
NITROGLYCERIN	38.96
POTASSIUM SULFATE	1.48
ETHYL CENTRALITE	1.01
TOTAL VOLATILES	1.00
LAMP BLACK	0.10

HEAT OF EXPLOSION: 1258 CAL PER G

BURNING RATE DATA:

TEMP °C	n	c 10^{-4}	PRESSURE LB PER SQ IN. 1000	2000	3000	4000
			BURNING RATE IN. PER SEC			
50	0.75	4.51	0.79	1.33	1.80	2.23
25	0.75	3.86	0.70	1.18	1.61	2.00
-25	0.81	2.00	0.53	0.93	1.29	1.63

$\bar{n}$	c'	T_l
0.78	0.649	229

FIGURE 3. The linear rate of burning as a function of pressure for a double-base propellant.

10.7.3 Burning Strand Method

This method is a new one and was developed at the Universities of Wisconsin and Minnesota.[1b] The apparatus consists of a strong steel vessel of approximately 300-cu cm capacity capable of withstanding a pressure of 25,000 psi. The lid of the bomb supports a framework on which a strand of powder about 5 in. long may be supported. Insulated leads through the lid connect with two fine wires which pass through the strand, one near the top and the other near the bottom. A coating of polyvinyl alcohol on the lateral surface of the strand prevents it from burning on any but the upper end

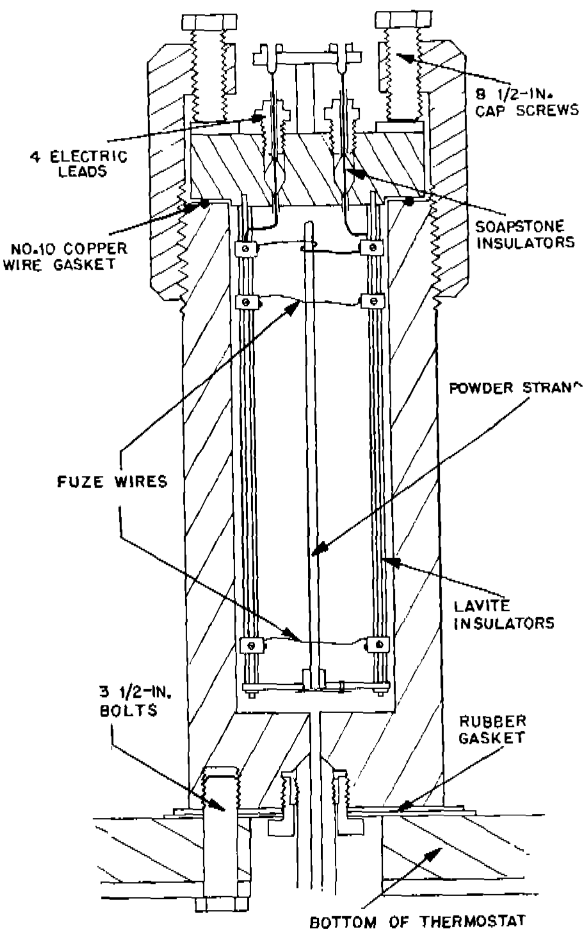


FIGURE 4. Diagram of apparatus for direct determination of the linear burning rate of a strand of powder.

surface. The strand and wire are placed in the bomb, the lid fastened tightly, and the whole immersed in a thermostat. The apparatus is illustrated in Figures 4 and 5. Inert gas is pumped into the bomb until the desired pressure is reached. When temperature equilibrium is attained, the strand is ignited at the upper end; as the flame passes each of the fine wires an electric circuit is broken, and the interval between the breaking of these circuits is recorded automatically. At the same time gas is exhausted from the bomb at a rate sufficient to keep the pressure constant. The length of powder between the two timing wires is accurately known, and hence the linear burning rate may be accurately measured.

This method of measuring burning rates has great advantages. It is direct, it requires very little powder for an experiment, and it is rapid. A variation of this method uses a bomb provided with a window so that the course of burning may be observed visually or by high-speed photography. By this method a large number of experimental powders have been investigated, and it should prove to be a valuable adjunct to any development or manufacturing program. It is most valuable for comparative measurements, since the radiation effects and the influence of the surrounding atmosphere of inert gas produce results that cannot be directly compared with those obtained when the powder is surrounded by a fairly thick layer of its own combustion products. Reports describing this technique and presenting the data on a series of double-base powders may be found among the final reports from the University of Minnesota[1b] and from the Allegany Ballistics Laboratory.[6]

10.8 SUMMARY OF EXPERIMENTAL RESULTS

10.8.1 Dependence of Burning Rate on Pressure

For most double-base powders and for some composite propellants, it was found that the burning rate data could be expressed within experimental error by either equation (1) or equation (2)—the linear or the exponential equations—between 200 and 500 psi. It seemed that equation (1) gave a better fit for some powders while equation (2) gave a better fit for others. It is certain, however, that both equations must be extended by the addition of another pressure dependent term if they are to fit data down to atmospheric pressure.[1d]

Propellants rich in nonexplosive plasticizers such as centralite or triacetin were found to give rate of burning-pressure curves that exhibited features hitherto unobserved.[3,5,9] The curves were S-shaped and even exhibited maxima. Powders L 4.8 and H–5 (see Table 2 of Chapter 13) both showed this

[redacted]

behavior. In this type of propellant there is, therefore, a region of pressure over which the burning rate varies very slightly. If equation (2) is fitted to the burning rate data in this range, the exponent n is found to be very small. It is emphasized that the region over which such a simple equation fits the complicated burning rate-pressure curve is small for these powders, but it is at least 1,000 psi. If, therefore, these propellants are burned in a rocket designed to develop an equilibrium pressure in the proper range, they possess all the advantages of a powder with a small n. The equilibrium pressures vary only slightly with temperature, surface, throat areas, etc.

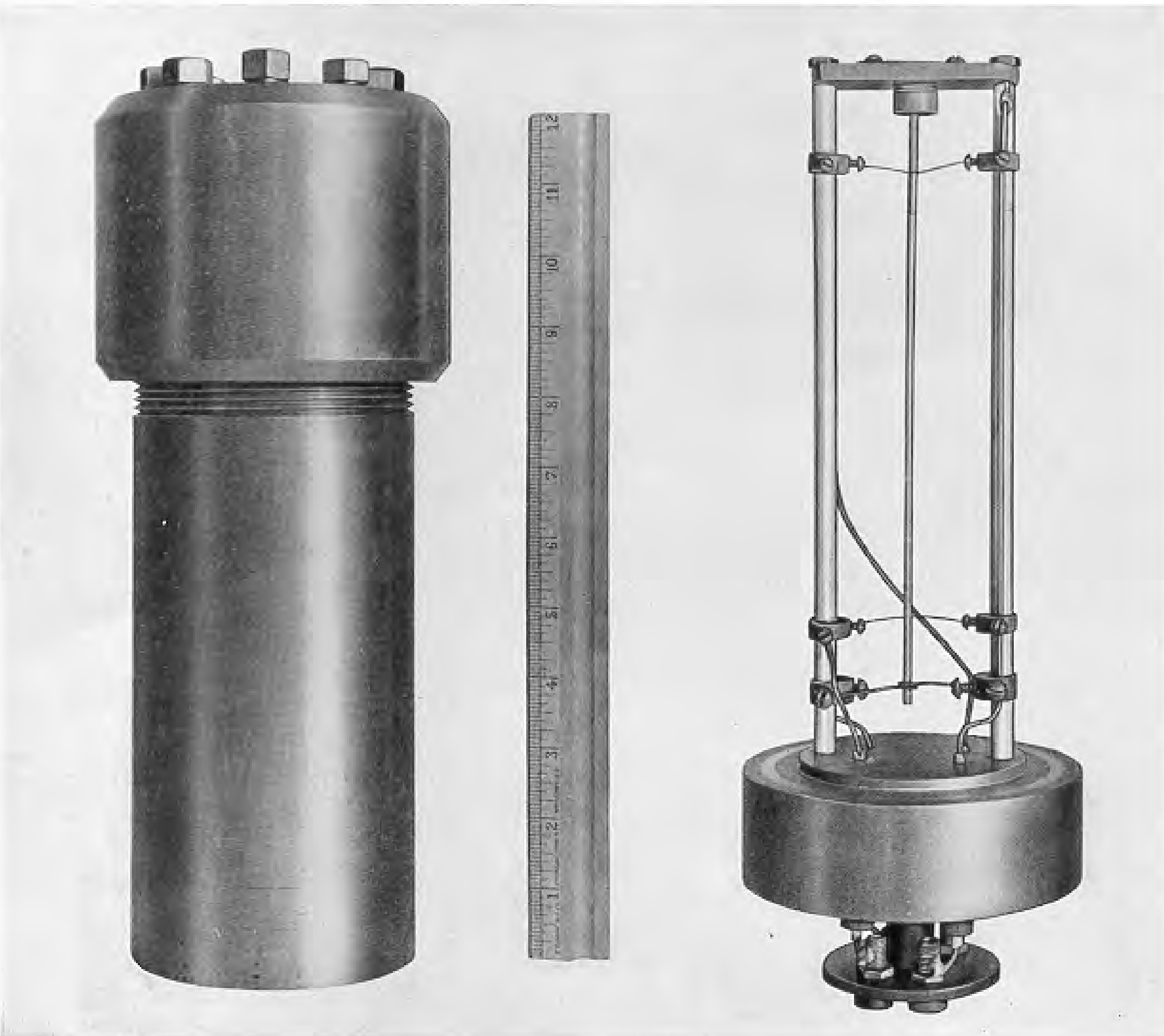


FIGURE 5. Photograph of apparatus for direct determination of linear burning rates.

In Table 1 the exponents and the corresponding pressure ranges are given for these powders and compared with those of ordinary double-base rocket propellants typified by the T–1 propellant which was available early in World War II. The explanation of the behavior of powders like L 4.8 or H–5 is not complete. It seems true, however, that at low pressures the nonexplosive plasticizers do not react completely with the explosive ingredient and hence the flame temperatures are higher than they would be if equilibrium were reached, because more carbon dioxide is formed. At higher pressures the nonexplosive plasticizers take more part in the reaction—carbon monoxide is formed in place of carbon

TABLE 1. Burning properties of various double-base propellants.

Propellant	Pressure exponent (n)	Pressure range (psi)	Temperature coefficient (percentage change in pressure per degree centigrade)
L 4.8	0.21	800–1,500	0.1
H–5	0.38	1,500–3,000	0.6
MJA	0.46	800–4,000	0.3
T–2	0.69	1,000–4,000	0.8
T–1	0.73	1,000–4,000	1.5

dioxide, and the flame temperature drops to the value expected on the basis of complete combustion. Since the rate of burning depends on the flame temperature, this explanation does give a picture which seems to be qualitatively correct.

10.8.2 Dependence of Rate of Burning on Temperature

It was found [15] that the burning rate of a propellant could be generally expressed as a function of temperature by an equation of the form

$$r = \frac{A}{T_1 - T} \tag{5}$$

or by combinations of (3) and (5) as

$$r = \frac{c'P^n}{T_1 - T}. \tag{6}$$

In these equations c' and n and T_1 are constants whereas T is the initial temperature of the powder. It will be seen that the larger T_1 is, the less will r change with T. A considerable variation in T_1 was found in the variety of powders studied, but no convincing generalizations were uncovered.

In actual rocket practice the variation of equilibrium pressure with temperature is a quantity of great significance. This quantity should be as small as possible to promote efficiency of design and constancy of thrust. It was found that reduction in n gave better practical results than increase in T_1. In the last column of Table 1, the "temperature coefficients" for the powders are given in terms of percentage change of equilibrium pressure with temperature under such conditions that the constants S, $(\rho - \rho_g)$, A_t, and C_D in equation (3) were held constant. It will be seen that for L 4.8 and MJA the temperature coefficients are much improved over that of the classical powders, as represented by T–1. Results for a number of other powders are in references 3, 9, and 18. This improvement did more than anything else to make smokeless rockets possible for the jet-assisted take-off of airplanes.

10.8.3 Dependence of Burning Rate on Chemical Composition

Examination of the burning rates of a fairly wide assortment of double-base powders showed that a plot of the burning rates at a given pressure against the heat of explosions of the powders (measured on a water liquid basis) could be expressed quite well by a straight line (see Figure 6). The higher the heat of explosion, the greater is the rate of burning under

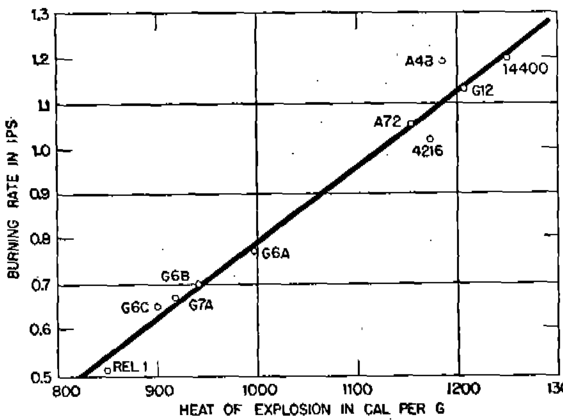


FIGURE 6. The linear rates of burning of a number of double-base powders as a function of their heats of explosion.

comparable conditions.[5,15,23] It is possible to calculate quite accurately the heat of explosion of a powder from a knowledge of its chemical composition and a table of constants characteristic of each ingredient.[23] We have, therefore, a means of determining approximately the burning rate of a powder if its composition is known, or, conversely, of specifying a composition of a powder to fulfill certain burning rate requirements. For the most accurate work, this relation must be supplemented by experimental determinations, but it is a good first approximation and proved of great value in designing propellants for new rockets. It was used with effect in designing the H–4 powder charge for the 115-mm aircraft rocket [14]—probably the most satisfactory rocket propellant yet developed—in the short space of a few weeks.

As a general rule, it was found that the slower the burning rate, i.e., the lower the heat of explosion, the smaller was the temperature coefficient for a powder. The important composition effect produced by the presence of large amounts of coolants such as triacetin has been already discussed under the dependence of burning rates on pressure.

10.9 INORGANIC SALTS

Inorganic salts such as potassium nitrate or potassium sulphate are well-known minor constituents of powders, but their effects on rocket propellants were not fully explored until recently. It is desirable to discuss these effects in two parts: first, the effect of small amounts of salts and, second, the effects of very large percentages of salts. When present in small amount (1 to 3 per cent), potassium salts modify the burning properties of the powder in several desirable ways: (1) they increase the ease of ignition, (2) they promote regular burning at low pressures, (3) they tend to reduce flash in the exhaust gases, and (4) they modify the course of the pressure-time curve. The flash-reducing properties were well demonstrated in the use of H–4 (T–2) powder both in the 115-mm aircraft rocket [14] and in certain modifications of the 4½-in. spinner rocket.[11] In general, the elimination of flash is brought about by cooling the exhaust gases to a sufficiently low temperature before they mix with the atmosphere. Two factors assist in this cooling process: the use of "cool" powder and the use of a large expansion ratio in the rocket nozzle. Both these effects, however, are helped by the addition of potassium salts to the powder. For example, it was found that in a given rocket a powder containing potassium nitrate was essentially flashless, whereas a powder of approximately the same heat of explosion but not containing potassium nitrate gave a brilliant flame in the exhaust.

The modification of the pressure-time curve by the introduction of potassium salts into a given powder composition was traced to the effect of radiation.[15] The presence of potassium salts in the hot gases from a powder increases the emissive power of the gases, and hence more radiation falls on the burning propellant per unit time, per unit thickness of radiating gas. If the opacity of the propellant grain is not sufficient to absorb all the radiation in a very thin outer layer, radiation will be absorbed in the body of the grain, and a rise in temperature will result. This will cause an increase in the rate of burning and a consequent rise of equilibrium pressure in the rocket. Since the amount of radiation falling on the propellant grain depends on the time, it will be seen that this provides a mechanism whereby the burning rate increases as the propellant is consumed, i.e., the powder burns progressively. In the ⅞-in. singly perforated stick granulation, it was found that the JPT powder without potassium nitrate gave regressive pressure-time curves, that is to say, the pressure rose to a maximum and then fell off slowly as the propellant was burned. This was due to the fact that these grains burned not only on the cylindrical surfaces but also on both ends, and consequently the area of the burning surface decreased during the reaction. The port area also increased during burning. When potassium nitrate was added to the composition, progressive pressure-time curves were obtained, the pressure rising steadily to the end of the burning.[5,15,16] It was found that the amount of progressivity in the burning of these grains could be controlled not only by the addition of potassium salts, but also by the addition of varying amounts of carbon black to the propellant in order to control its absorption coefficients for radiation. This phenomenon is discussed in detail in references 5 and 24. It should be emphasized here, however, that in the design and manufacture of a first class double-base rocket propellant considerable care should be given to specifying the proper salt content and carbon black content to give the desired type of pressure-time curve. The hotter the powder, the more attention to these details is required.

When large amounts of inorganic salts were incorporated in double-base powder, together with somewhat smaller amounts of carbon or other solid reducing agent, entirely new effects were seen, the most important being a marked decrease in the pressure exponent of the powder. This phenomenon is exemplified in solvent-extruded composite propellants which consisted of a nitroglycerin-nitrocellulose powder in which was incorporated upwards of 50 per cent of potassium perchlorate or potassium nitrate and several per cent of carbon. A typical composition of this propellant is given in Table 6 of Chapter 13. A large and successful development program to make these powders was carried out by Division 8, NDRC. Part of this is described in joint Division 3 and 8 final reports.[17,18] For a full account the Summary Technical Report of Division 8 should be consulted.

10.10 BURNING RATES AND RADIATION

Some effects of radiation on the burning of rocket propellants have been outlined in the preceding paragraphs. Generally speaking, the radiation from the hot powder gas influences the burning of the powder grain by penetrating below the reacting layer and causing a rise of temperature which in turn increases the burning rate. The magnitude of this rise of temperature increases with the total emissive power, itself a function of the temperature,[8] and the thickness of the hot gas surrounding the grain. It also depends on the absorption coefficient of the powder itself in the appropriate regions of the spectrum. The effect of radiation in causing the progressive burning of rocket powders was investigated experimentally and theoretically by consideration of the above-mentioned factors, and the agreement between the results of these two lines of attack indicates that the phenomenon is fairly well understood.[7,8] An interesting example of the influence of radiation was noted in the development of grains with long burning times (10 seconds) for jet-assisted take-off work.[9]

A very drastic example of the effect of radiation on the burning of powders of high calorific value was discovered early in World War II. The effect was so serious that for a while it was doubted whether double-base powder could be used as a reliable rocket propellant. In the early days of rocket development by Section H, Division 3, NDRC, considerable trouble was encountered from the presence of cracks, fissures, or other flaws in the powder grains. When the grains burned, the flame entered these fissures, greatly increasing the burning surface area over that predicted and causing a large increase in internal pressure with consequent violent disruption of the rocket. Research in the manufacturing process resulted in the overcoming of this difficulty, but the possibility of fissures being present in the propellant grains was so serious that a powder was developed which was translucent enough to allow positive visual inspection of the grains for fissures or flaws. This powder had a composition similar to JPT in Table 1 of Chapter 13, had a high heat of explosion, and was made in grains that were absolutely flawless with a negligible percentage of rejects. Nevertheless, rockets continued to blow up. A technique for extinguishing powder grains before the burning was complete revealed that a number of the translucent grains developed numerous fissures during burning, and hence the area of the burning surfaces increased with disastrous effects. Figure 7 shows the type of phenomena encountered with $\frac{7}{8}$-in. JPT powder. Intensive study showed that the effect was due to radiation

FIGURE 7. Radiation fissuring of a hot double-base powder. Comparison of partially burned and unburned grains.

from the hot powder gases penetrating the powder and causing strong local heating when absorbed by a trace of dirt or an accidental region of high absorbing power. The remedy consisted in introducing sufficient coloring matter into the powder to absorb the radiation almost completely in the outer layers.[1,3,14,16] It was found with all hot powders (heats of explosion on a water liquid basis greater than 1,000 calories per gram) that stable burning in rocket motors is possible only when the powder is made sufficiently opaque to radiation. At the California Institute of Technology the same difficulties were encountered with a powder very similar in composition to JPN (see Table 1 of Chapter 13) and solved in the same way.

10.11 THEORETICAL WORK

As may be well expected, reaction of double-base powder to produce oxides of carbon, hydrogen, water, and nitrogen is an extremely complex process. It has been attacked theoretically both in this country and in England, and it is safe to say that a theory is now worked out to a point where the general processes are qualitatively understood and some quantitative predictions can be made. The theory is not in a shape where definite simple generalizations can be made.

Very briefly, the theory of burning at moderate pressures (of the order of 10,000 to 15,000 psi) assumes that the burning reaction takes place in three stages: a first-order monomolecular decomposition which takes place just below the burning surface, a second-order monomolecular reaction which takes place in the gas phase close to the burning surface, and a branched chain reaction which takes place in the gas phase at somewhat greater distance from the burning surface. The second stage has been referred to as the dark-zone reaction and the third stage as the luminous or flame reaction. The overall rate-controlling step is assumed to be the surface reaction, which in all probability is an exothermic decomposition reaction involving the formation of nitrogen dioxide. The rate of this reaction depends chiefly on the temperature of the powder very close to the reacting surface, and this temperature in turn depends on the rate of heat transfer from the hot gas phase back to the surface. In the steady state there is a steep temperature gradient from the powder surface to the flame zone and a steady heat flow across any cross section between the powder surface and the flame. As the pressure increases, the reaction zones become narrower and approach more closely to the surface, thus increasing the temperature gradient, the rate of heat transfer back to the surface, and the overall burning rate. The dark-zone reaction probably involves the production of aldehydic substances and nitrogen oxide. This reaction probably contributes about one-half of the total heat and always takes place. The flame reaction involves the burning of the aldehydic substances and the nitrogen oxide to give carbon monoxide, carbon dioxide, water, etc. It always takes place at high pressures but may fail to go at low pressures. It is interesting to note that failure of the flame reaction is apparently closely connected with the irregular burning of propellants in rockets at low pressures and low temperatures. Attempts to improve this failing on the part of double-base propellants have centered around the use of inorganic substances to catalyze the flame reaction. This theory, although undoubtedly an oversimplification of the actual mechanism, is able to account roughly for the observed temperature and pressure dependence of the burning rate, and reasonable activation energies for the various stages can be postulated. The fundamental mathematical treatment developed by Boys and Corner in England is the basis of most of the theoretical work. Further details of the theory of the burning of powders may be found in references 1c, 4, 19, 20, and 21.

Chapter 11

STRUCTURAL PROBLEMS

By *R. E. Gibson*

11.1 INTRODUCTION

In order to fulfill its purpose, a solid rocket propellant must be formed into a given size and shape, must be supported adequately in the rocket motor, and must possess mechanical properties good enough to withstand the stresses imposed upon it under firing conditions and during handling and storage. During World War II all agencies engaged in rocket development expended a considerable amount of effort in solving problems connected with the design of propellant charges, with the methods of making these charges, with the stresses set up in propellants under working conditions, and with the development of propellants whose physical properties were adequate to withstand these stresses. Both theory and experiment were applied to these problems. As a general result, it may be stated that the empirical work gave solutions to the more immediate practical problems, but a satisfactory theory of the solid state of colloidal propellants is still to be written.

This chapter will deal mainly with work done by the laboratories associated with Section H, Division 3, NDRC, and will be concerned with double-base powders. Much work on the physical properties of composite propellants was done by Division 8, NDRC, and the reader is referred to the final reports of that Division for details.

Under the general heading of structural problems may be included the following subjects: (1) charge design; (2) granulation; (3) physical properties of propellants.

11.2 CHARGE DESIGN

The propellant charge in a rocket may consist of one or more "grains" of powder. The individual grains may weigh anything from a fraction of an ounce to several hundred pounds. Each grain, however, must be so made that its burning surface will remain essentially constant during the whole burning period. The reasons for this prime requirement were brought out in Chapter 10 of this report. The number, size, and shape of propellant grains to be used in any rocket depend on the performance required of the rocket and cannot be discussed in any condensed form. In Chapter 13 of this report, some of the general principles as they relate to existent propellants are delineated. An excellent discussion of the problem of designing a propellant charge for a modern high-velocity rocket is given in reference 22, where several generalizations of

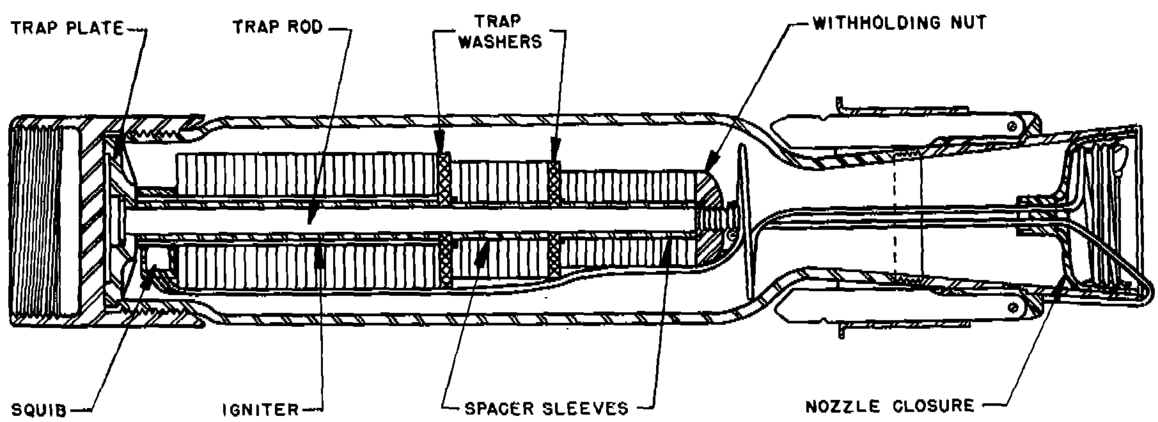


FIGURE 1. Diagrammatic sketch of rocket burning laminated charge of powder disks.

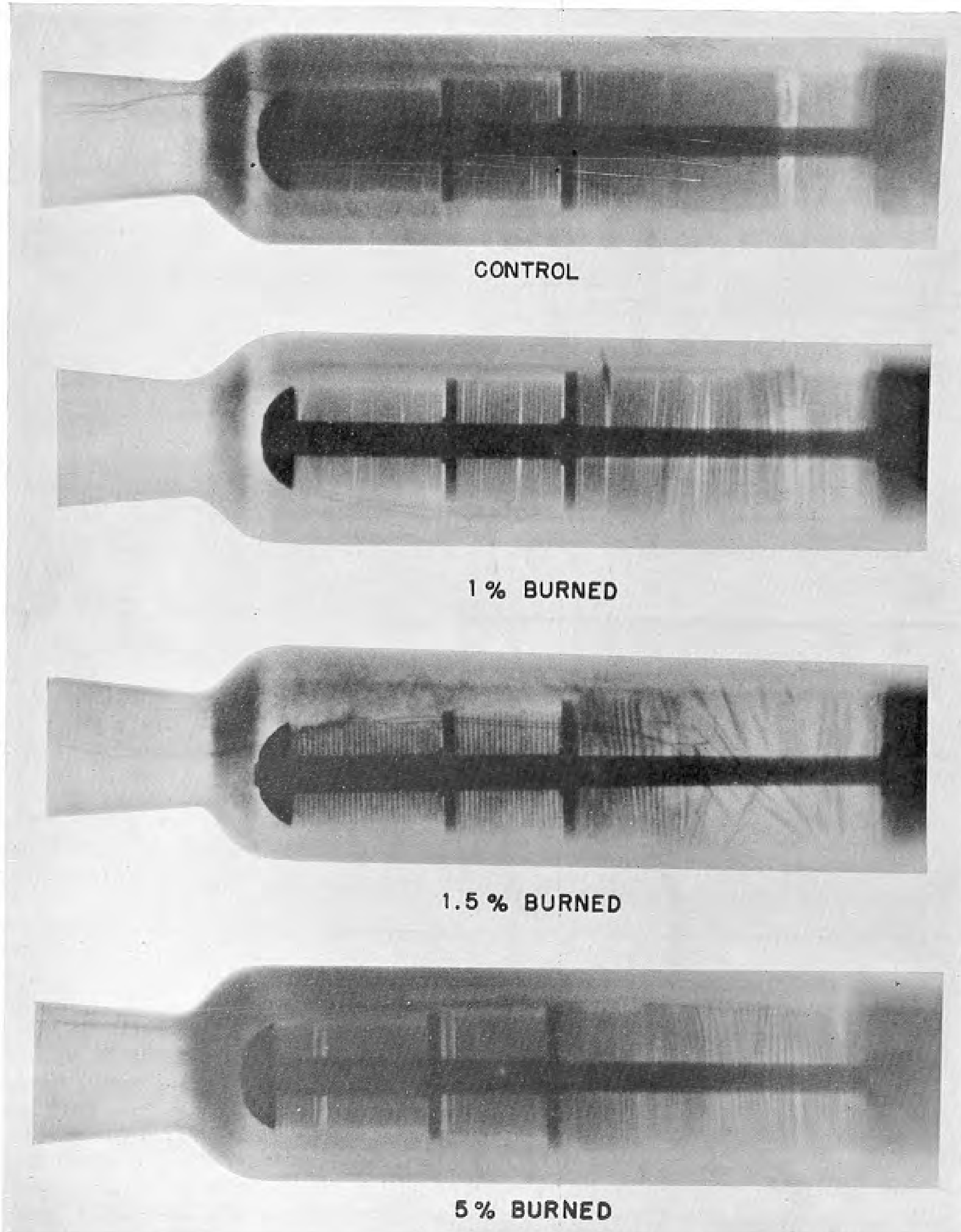


FIGURE 2. X-ray spark photograph of propellant burning in superbazooka rocket. Note disk powder grains.

wide application are made. In this section we shall merely refer to a few types of granulations that were used and name the reports in which they are described.

Where high accuracy of the rocket is needed, it is necessary to guide the projectile until the burning is finished. This was found to be a requirement in the bazooka rocket, and, in order to keep the

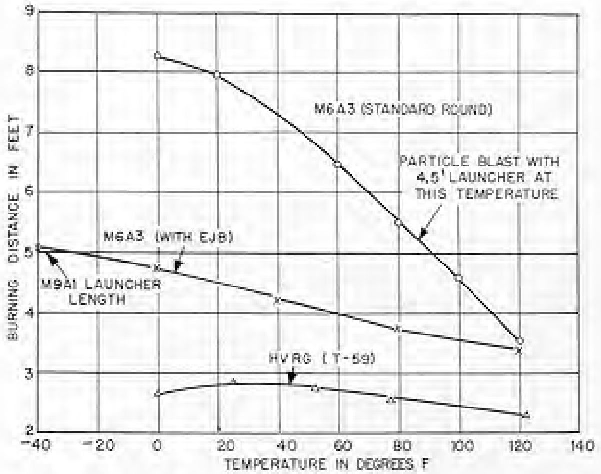


FIGURE 3. Effect of temperature on burning distance of superbazooka rocket.

launcher down to a convenient length, a very short burning time was essential. Thin-web grains of fast-burning powders were used, and, in order to have sufficient powder to supply the necessary momentum, the charge design called for a number of singly perforated grains.[1-4]

In the superbazooka (T–59 rocket) which had a higher velocity and greater payload than the ordinary bazooka, an entirely new type of charge was designed. This charge consisted of a number of disks of sheet powder having a web of approximately 0.06 in. The disks were perforated and held in place by a steel rod passing through the perforations and anchored at the fore end. (See Figure 1.) Adequate port area was provided by progressive reduction of the size of the disks, the smallest disk being nearest to the nozzle. Figure 2 illustrates this charge. The photographs were taken by high-speed X-ray photography and actually show various stages in the burning of the charge.[5] This charge had a very short burning time of the order of 20 milliseconds—so short indeed that equilibrium pressure was never reached, and hence the effect of temperature on burning time and maximum pressure was reduced to a very low value. Figure 3 shows the practical advantages of this charge by illustrating the effect of temperature on the burning distance. The curve marked HVRG refers to the rocket using the charge just described. It will be seen that the burning distance of the discharge varies very little with temperature when compared with other experimental rounds and extremely little when compared with the standard M6A3 round. For fuller details the reader is referred to the complete reports.[5,6]

In the M–8 rocket and its modifications (see Figure 4), the 115-mm aircraft rocket and its modifications, and the 4.5-in. spinner rockets, the propellant charges consisted of a number of singly perforated cylindrical grains, all the surfaces of which were allowed to burn. The grains were supported by wires passing through their perforations and connected together to form a cage-like trap. By

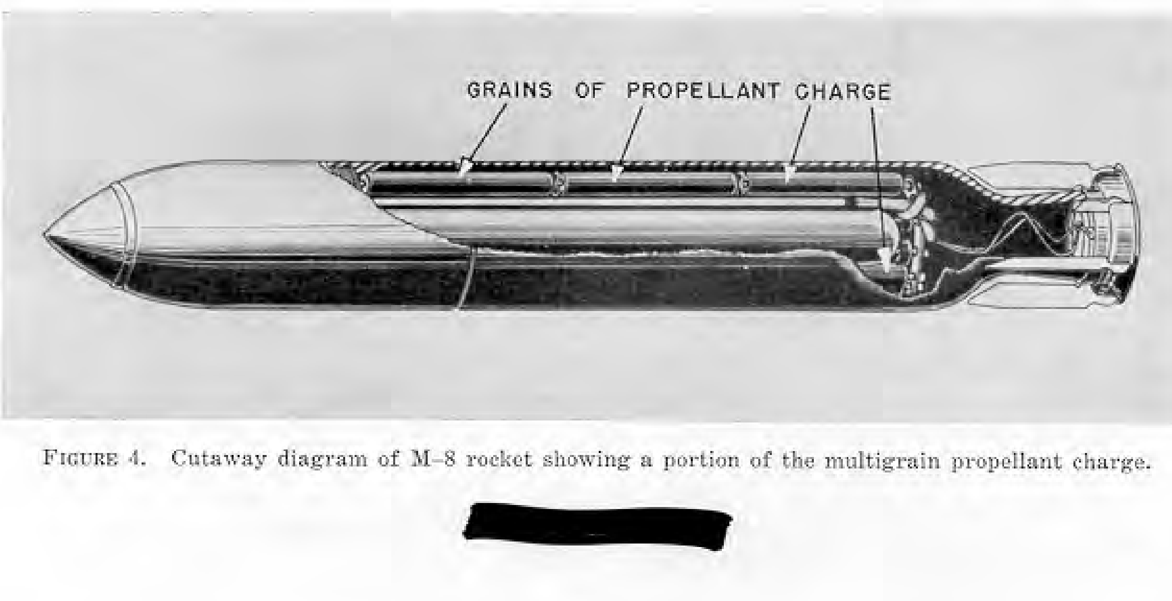


FIGURE 4. Cutaway diagram of M–8 rocket showing a portion of the multigrain propellant charge.

[redacted]

control of powder compositions a wide variation in burning time may be realized with this type of charge even though the maximum web thickness is not greater than half an inch. The ease of manufacture of this type of grain and the fact that, since a number of grains are used, statistical methods may be applied in specifications and inspection constitute the chief advantages of this design. The disadvantages arise chiefly from the problem of assembling the propellant charge in the round. The details of the design and performance of this type of propellant charge are given in reports dealing with the weapons in which it was used.[6–10]

The technique of designing propellant charges consisting of large single grains of powder, either in the form of singly perforated cylinders or columns of cruciform cross section, was developed by the British and applied in this country very successfully by Section L, Division 3, at California Institute of Technology whose final reports should be consulted for a complete account of the subject. (See also Part II of this volume.) Considerable application of this type of charge was found in rockets used for towing or pushing demolition charges, a development carried out jointly by Allegany Ballistics Laboratory and the Corps of Engineers.[11–15] An example of a motor used for towing a detonating cable is shown in Figure 5. With a fairly slow powder, burning times up to 5 seconds may be readily obtained with this type of charge.

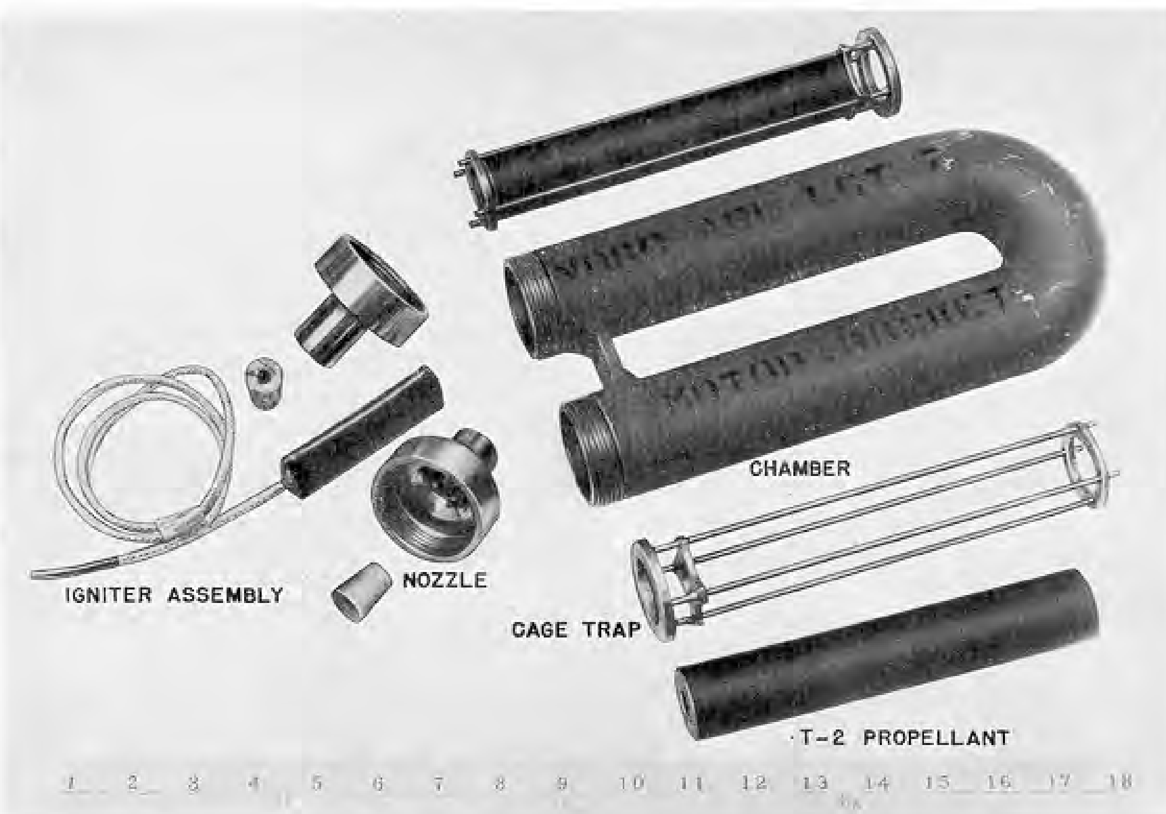


FIGURE 5. Rocket for towing detonating cable. Note form of propellant charge and its support.

11.3 INHIBITED GRAINS

The possibility of preventing double-base powder grains from burning on certain surfaces by coating these surfaces with an adherent layer of noninflammable plastic has greatly extended the possibilities of propellant charge design. Certain compositions of ethyl cellulose and of cellulose acetate have been found suitable as inhibiting coatings, but a large number of other agents have been examined.[16,17] It has been found quite feasible to restrict burning of double-base powder reliably by these means, but the problem of diffusion of nitroglycerin from the powder to the coating or of plasticizer from the coating to the powder is one which can be solved completely only by long-time surveillance tests. At present ethyl cellulose compositions give least trouble from this source.

Solid cylindrical grains coated with plastic on the cylindrical surface and, therefore, restricted to burning on the ends have a neutral-burning geometry and give charges of very long burning time. This type of charge was investigated extensively at Allegany Ballistics Laboratory (following the lead of the British workers) in connection with the development of a JATO unit and of a device for pressurizing a one-shot portable flame thrower. The reader is referred to the original reports for details,[19–21] but Figure 6 gives a general idea of the type of restricted burning grains used for the propellant charge in the JATO.

Another important type of rocket propellant charge that depends on restriction of burning for its action is that using a single grain, burning only in the perforation and on the end adjacent to the nozzle. This charge possesses a great advantage in that the powder itself insulates the walls of the chamber from the action of the hot gases and permits the use of light alloys in the fabrication of the body of the rocket motor. It offers the most promising possibilities for developing high-velocity artillery rockets—possibilities which were realized in the scale model of the Vicar, a rocket which carried a useful payload at a velocity exceeding 2,600 fps.[22]

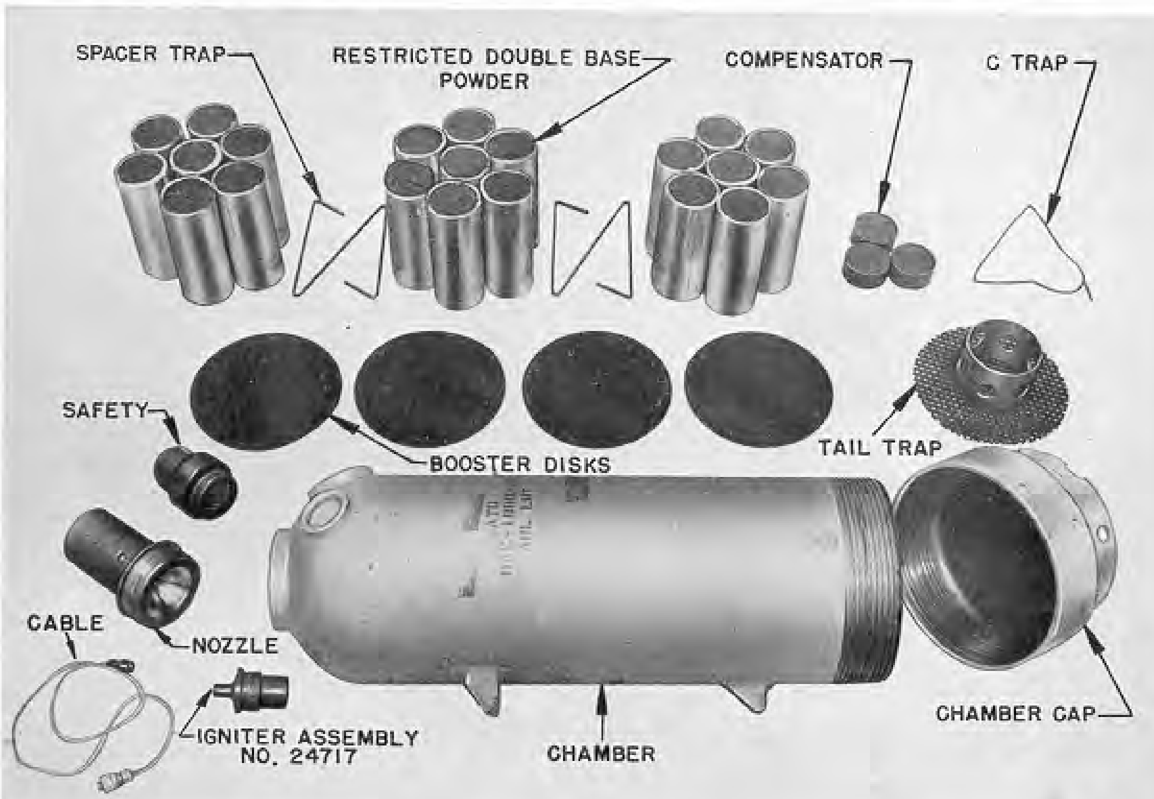


FIGURE 6. Photograph of disassembled JATO unit showing propellant charge and support.

In this charge, the outer surface and the fore end of the grain are coated with plastic and made to fit snugly in the rocket motor. Neutral burning is achieved by making the perforation star-shaped in cross section so that its perimeter is equal to the outside perimeter of the cylinder. Such a charge is illustrated in Figure 7 and its development and performance are described in references 22 and 23. During 1946 and 1947 Allegany Ballistics Laboratory developed this charge to an advanced stage in the "Deacon Rocket," which carries approximately 100 lb of propellant, 50 lb of payload, and 50 lb of deadload, and attains a velocity exceeding 4,000 fps.

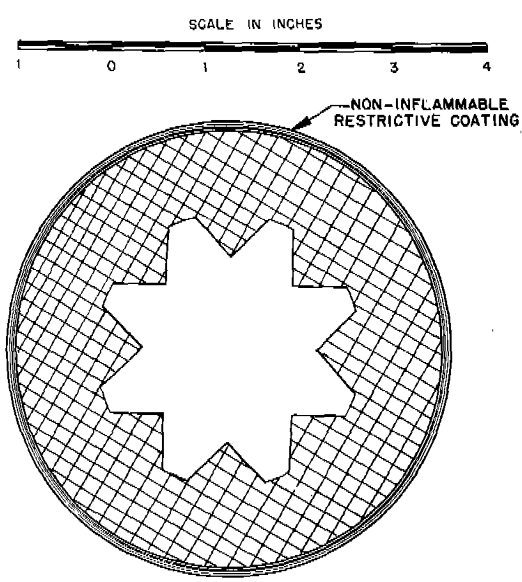


FIGURE 7. Diagram of internal-burning charge showing star-shaped perforation.

11.4 GRANULATION

Under this heading come all the problems concerned with the preparation of reliable grains of propellant in the proper shapes, sizes, and types to meet the requirements imposed by the applications.

The general methods used in granulating double-base powder are outlined in Chapter 13. Here we shall merely refer to one or two outstanding problems associated with each method.

In preparing grains by solvent extrusion, control of dimensions and the prevention of warping on drying were difficult problems. They were solved largely by the efforts of the staff of the Hercules Powder Company at Radford Ordnance Works, and the reader is referred to reports from this organization[a] for a complete account of the work. Fissures appearing in the grains after extrusion also gave difficulty and were never entirely overcome—the use of carbon dioxide to replace air in the presses was suggested by the University of Wisconsin group and gave considerable promise. The greater solubility of carbon dioxide in acetone was the basis of this proposal.[24]

A considerable amount of work was expended on a study of the "dry" extrusion of solventless powder. Since the development and manufacturing phases of this subject were thoroughly studied[b] elsewhere,[25] most attention was given to experimental work, die design, studies of flow of plastic through dies, effect of composition on the extrudability of powder, influence of pressure, temperature, and rate of extrusion on the finished product. These studies were closely linked with examination of the product under ballistic conditions.[26]

In connection with the developments outlined in the previous section, extensive studies were made of methods of restricting the burning surface of propellant grains. This work was based on the very important developments made by the British workers and indeed was chiefly aimed at adapting their methods to powders, plastics, and adhesives available in this country. Several satisfactory methods of restricting powders were developed, and extensive studies were made of the effect of stress set up by temperature changes and by shock during handling, transportation, and firing conditions. This work has continued at Allegany Ballistics Laboratory under the Hercules Powder Company and has been extended and improved. Details of the status at the end of 1945 are to be found in reference 16.

A very significant advance in granulation technique was made by Division 8, NDRC, in the development of double-base powder which could be cast in a fluid state and set up to rigid grains of good mechanical properties by storage under proper temperature conditions. Details of this work can be found by reference to the Summary Technical Report of Division 8.

11.5 PHYSICAL PROPERTIES OF ROCKET PROPELLANTS

Under firing conditions a propellant grain is supported by a suitable trap and is acted on by forces due to setback, differential pressure, and igniter shock. The stresses set up are complicated, and the definition of those properties whose quantitative expression indicates the ability of a grain to stand the stresses is even more complicated. Some work

[a] To the Ordnance Department.

[b] See Chapter 7.

was done on the quantitative determination of the stresses set up in propellant grains under firing conditions,[27] but it is emphasized that a great deal of experimental work is still needed in this field. In parallel, studies were made of the elastic properties of double-base propellants, such as Young's modulus and the coefficient of thermal expansion, and of the resistance of the powders themselves to stresses applied in different ways and at different rates with the intention of producing mechanical rupture. Several pieces of apparatus were devised especially to carry out these experiments, particularly to duplicate the rates of application of load presumed to exist in actual rockets. Compressive strength, impact values, tensile strength, resistance to indentation were among the qualities measured. The results are not susceptible of generalization in a condensed form, and the reader is referred to the original reports for the results.[28]

A very important method of determining the ability of a powder to withstand the stresses set up under firing conditions is a comparison of the pressure-time curves obtained under static and flight conditions. By careful measurements of the velocity of a rocket during burning it is possible to calculate an acceleration-time curve and with the help of auxiliary data to convert this into a pressure-time curve. If any discrepancies between the static and the flight-pressure-time curves are noted, it is well to examine carefully the physical properties of the propellant and the nature of its support in the motor, since trouble that might develop to serious proportions is indicated long before it shows itself by disastrous effects.[8,9]

Chapter 12

INTERIOR BALLISTICS PROBLEMS

By *F. T. McClure*

12.1 SIMPLE BALLISTICS

As pointed out in chapter 10, the burning law for most propellants can be represented, to a first approximation, in the form[a]

$$r = cP^n, \tag{1}$$

where c is a constant characteristic of the propellant and initial charge temperature, and n is a constant characteristic of the propellant.[b] Detailed discussions of the experimental studies of the burning laws for powders are available in references 1, 2, 3, and 4.

With this form of the burning law, simple considerations of the balance of gas production and discharge lead to the expression

$$P = \left[\frac{Sc(\rho - \rho_g)}{A_t C_D}\right]^{1/(1-n)}, \tag{2}$$

for the equilibrium operating pressure of the rocket motor. In this equation, S is the powder surface area, c and n are the constants of the burning law, A_t is the throat area of the nozzle, C_D is the discharge coefficient of the gas, ρ is the density of the powder, and ρ_g is the density of the gas in the motor chamber (usually quite small compared to ρ).

The significance of equation (2) as an illustration of the influence of large values of n in magnifying the effects on P of small changes in S or c is discussed in Chapter 10.

Although equations (1) and (2) represent satisfactory approximations in the case of motors which have relatively large cross-sectional areas free of powder, they neglect effects which become progressively more important as cross-sectional area of the motor chamber is more completely filled with powder. Thus, in the design of modern, lightweight, high-performance rocket motors, more detailed knowledge of the burning law and equilibrium pressure law is necessary in order to include the important influence of the so-called "throat-to-port ratio." The throat-to-port ratio, A_t/A_p, is the ratio of throat area to the cross-sectional area of the free space in that part of the motor which contains the propellant powder (i.e., the so-called "port area").

[a] See Section 5.3.3 for another form of this equation.

[b] The use of the symbol n in this chapter as the exponent of the burning law must not be confused with its use in Chapter 9 as the inverse of the molecular weight of the gas.

12.2 INFLUENCE OF THROAT-TO-PORT RATIO ON THE DISCHARGE COEFFICIENT

Theoretical consideration of the flow of gas in the channels along the sides of the propellant grains leads to the conclusion that the discharge coefficient will have a small dependence on the throat-to-port ratio because of the pressure drop and associated gas velocity in the propellant channels. Detailed analysis is carried out in Appendix 6 of reference 5, leading to the conclusion that the effect may be represented with reasonable accuracy by an equation of the form

$$C_D' = C_D[1 - \phi(A_t/A_p)^2], \tag{3}$$

where C_D' is the effective discharge coefficient, C_D is the ideal theoretical discharge coefficient, and ϕ is a weak function of the γ[c] of the gas, running from 0.21 at $\gamma = 1.2$ to about 0.23 at $\gamma = 1.4$.

12.3 INFLUENCE OF THE THROAT-TO-PORT RATIO ON THE BURNING LAW

12.3.1 Pressure Drop

The pressure drop along the propellant channel results in a "space average" pressure which is slightly less than the head end pressure in the motor. The space average burning rate corresponding to this space average pressure determines the rate of gas production in the motor. Correction of the burning law for this "pressure drop" effect[6,7] leads to a burning law of the form

$$r = cP_0^n[1 - \tfrac{2}{3}\phi(A_t/A_p)^2]^n, \tag{4}$$

[c] Defined in Section 9.2.

where $\bar{r}$ is the space average burning rate, and P_0 is the head end pressure, and the other symbols have their previous significance.

12.3.2 Erosive Burning[d]

There is still another effect of the flow in the propellant channel which is more important than those mentioned above. The rate of burning of the propellant depends on the velocity as well as the pressure of the gases flowing over its surface. Higher velocities produce higher rates of burning. This is made strikingly clear by the observed "tapering down to the rear" of partially burned grains, in direct contradiction to the effect to be expected if pressure alone were the sole determining factor in the burning law. A relatively complete and detailed experimental study of this problem of erosive burning is presented in reference 6, leading to the clear conclusion that the basic burning law is much better represented by the form

$$r = cP^n(1 + kv) \tag{5}$$

than by equation (1). In equation (5) v is the velocity of the gas, and k is the so-called "erosion constant." The constant k must be determined experimentally for the propellant, and reference 6 discusses the methods of accomplishing this end.

Again, in ballistic calculations the space average burning rate is the quantity of importance, and this can be expressed [6,7] in the form

$$\bar{r} = cP_0{}^n[1 - \tfrac{2}{3}\phi(A_t/A_p)^2]^n[1 + 0.5k_2(A_t/A_p)], \tag{6}$$

where k_2, the "erosion constant in terms of throat-to-port ratio," can be calculated from the erosion constant, k, and the thermodynamic properties of the propellant gas.

12.4 BALLISTIC EQUATION INCLUDING THE THROAT-TO-PORT RATIO

When the throat-to-port effects discussed in the previous sections are included in the calculation of equilibrium pressures, the equation

$$P_0 = \left\{ \frac{c(\rho - \rho_g)S[1 - \frac{2}{3}\phi(A_t/A_p)^2]^n[1 + 0.5k_2(A_t/A_p)]}{A_t C_D[1 - \phi(A_t/A_p)^2]} \right\}^{1/(1-n)} \tag{7}$$

is obtained for the equilibrium pressure at the head end of the rocket motor. This equation is discussed briefly in reference 6 and in more detail, especially with respect to rocket design, in reference 7. The experimental design work described in reference 8 verifies the essential correctness of equation (7) and establishes it as a basic equation in the design of solid fuel rocket motors. Current reports from the Allegany Ballistics Laboratory (now operated by the Hercules Powder Company under contract with the Navy) lend ample support to this claim.

Equation (7) clearly demonstrates the influence of the throat-to-port ratio on the pressure obtained in a rocket. Since the port opens up as the powder burns away, this throat-to-port effect decreases with time (largely due to the decrease in erosive burning). The effect is thus a *regressive* one, and to obtain constant pressure operation (necessary for light-walled motors) it is necessary to design the grain with a *progressive surface* to counterbalance the throat-to-port effect. The importance of equation (7) in determining the desired surface progression is obvious. It is also clear that, in order to have adequate information on which to design modern solid fuel rockets, it is necessary to know the *three* burning law constants, c, n, and k, as well as the thermodynamic properties of the powder gas.

Although equation (7) takes account of the most important factors which influence the equilibrium pressure of a rocket motor, there are other factors which may produce marked effects under more specialized conditions. Two of these factors are discussed briefly in Sections 12.5 and 12.6.

12.5 RADIATION

The radiation from the hot powder gases also affects the burning rate of the powder, and under special conditions may produce extremely large effects. The general problem of radiation in the rocket chamber and its influence on the burning of the powder is discussed in some detail in references 9 and 10.

Although qualitative and sometimes semiquantitative treatment of special radiation effects (such as fissuring, end pressure peaks, and influences of wide gas channels and metal walls on burning rate) have been possible, complete integration of the radiation phenomenon into the ballistics system has not yet been, as far as the author knows, successfully accom-

[d] See also Section 5.3.2.

[redacted]

plished. Present knowledge, however, indicates that such integration is possible if the time and labor are made available to do the job.

12.6 RESONANCE EFFECT

An unusual effect, not yet completely explained, is the so-called "resonance effect." This phenomenon results in the appearance of greatly increased pressures part way along in the burning. These peak pressures frequently last for only a short period, and then the pressure drops again to the normal equilibrium value and the burning process continues as though nothing unusual had happened. The resonance phenomenon is apparently highly specific with respect to powder and motor geometry and also operating conditions. Although the phenomenon can generally be prevented by "breaking up" the geometry (such as by putting a metal rod down the perforation of a grain), there appears to be no way as yet of predicting whether or not a given motor design will display the phenomenon. This would appear to be a realm in which considerable advance in knowledge is *highly desirable.*

12.7 DRAG OF THE GAS STREAM ON THE PROPELLANT

Two factors contributing to the forces tending to cause mechanical failure of the propellant in a rocket motor are the acceleration forces, and the drag of the flowing gases on the propellant charge. The former is easily calculated, but the latter is more involved.

Reference 11 provides a simple theory of the drag of the gases on the charge and a limited experimental verification of proposed formulas, which give the drag as a function of the cross-sectional area of the charge and the throat-to-port ratio. Further experimental study of these and other forces on the propellant charge are of definite interest to the future design of solid fuel rockets.

12.8 HEAT TRANSFER TO THE MOTOR WALLS

The problem of the heat transfer from the hot gases to the motor walls is important because of the consequences in reducing the strength of the metal.

Reference 12 contains a theoretical discussion of the heat transfer problem, and reference 13 considers some of the experimental problems associated with making significant measurements. (See also Part II and Chapter 23 of this volume.)

12.9 NONSTEADY-STATE ROCKETS

The ballistic laws discussed in the earlier sections of this chapter apply to rockets which operate under equilibrium conditions. In very special cases, it may be advantageous to operate a rocket motor in which the pressure is limited only by the complete consumption of the propellant. Design of charges for this sort of application is discussed in Section 11.2 hereof. Such motors, however, appear to have a very limited application. Their ballistics and the design of such a motor are discussed in detail in reference 14.

12.10 SPECIFICATIONS AND TESTING OF PROPELLANTS

The problem of setting down specifications and control testing procedures which will assure that a mass-produced propellant will behave as intended is a difficult one. It can only be approached from the standpoint of a basic knowledge of rocket ballistics. This approach was explored during World War II, and considerable success was achieved in formulating rational specifications based on scientific knowledge. Reference 15 discusses this problem in considerable detail, using as specific examples powders which were standardized during World War II.

[redacted]

Chapter 13

PROPERTIES OF ROCKET PROPELLANTS AVAILABLE OR DEVELOPED DURING WORLD WAR II

By *R. E. Gibson*

13.1 INTRODUCTION

In this chapter we shall give a description of the properties of propellants which were either available or developed between 1940 and 1945. In order to make the chapter as self-contained as possible, we shall first gather together the definitions of quantities significant in the use of rocket propellants, then discuss the various classes of propellants in terms of these quantities, and, in the case of each class, present a table summarizing the compositions of representative members. In discussing the properties of these various powders, an attempt is made to bring out considerations which are of significance in the design of new rockets. The chapter ends with a short section suggesting lines along which research and development work in the field of rocket propellants may proceed in the future. The substance of this chapter appears as part of one of the Allegany Ballistics Laboratory final reports.[1] The report was originally written by the author as the technical section of a final report from the Rocket Propellant Panel to the Joint Committee on New Weapons and Equipment. It, therefore, includes the work of a large number of agencies and is wider in scope than the preceding chapters.

13.2 SIGNIFICANT CHARACTERISTICS OF SOLID ROCKET PROPELLANTS

13.2.1 Specific Impulse and Effective Gas Velocity

The thrust imparted to a rocket when unit mass of powder gas is discharged per second[a] is a quantity which is of great interest in rocket design and which depends primarily on the thermodynamics of the propellant gas, as modified slightly by heat losses, and secondarily on the expansion ratio of the rocket nozzle. In ordinary units it may be expressed as [lb(force) × seconds] per [lb(mass)] and is called the specific impulse. It will be seen that the specific impulse multiplied by the mass of powder burned gives the total impulse, that is to say, the momentum, given to the rocket. If the thrust imparted to the rocket per unit mass of powder discharged per second is expressed in common velocity units by converting lb(force) to lb(mass), it is called the "effective gas velocity" of the propellant. In ordinary units effective gas velocity = 32.2 × specific impulse (32.2 being the acceleration of gravity).

One of the problems in the development of rocket propellant is the search for propellants of greater specific impulse, since, it will be noted, the velocity increase of a jet-propelled device of given weight and carrying a given weight of propellant is almost directly proportional to the specific impulse of the propellant.

13.2.2 Burning Time

The total momentum (mass × velocity) given to a rocket device may be conveniently regarded as the product of the thrust multiplied by the time the thrust is applied (more rigorously, the integral of the thrust multiplied by the time). The accelerations and the mass of gas discharged per second, a measure of the blast of the jet, are both proportional to the thrust, and either may impose upper limits on the allowable value of the thrust. The time of application of the thrust, i.e., the burning time of the propellant, is, therefore, an important engineering variable. The burning time of a rocket propellant charge depends on two quantities: (1) the *linear burning rate of the propellant* and (2) the *distance the burning surface must move* as the flame consumes the propellant. This latter is commonly referred to in terms of the web thickness of the powder grains.

The *linear burning rate of a propellant* depends

[a] The mass of gas discharged per second is equal to the mass of powder burned per second when a steady state is reached in the rocket.

primarily on its composition, its temperature, and the pressure of the gas over it, and secondarily on the radiation falling on it and the rate of gas flow over its surface. It has been discussed fully in Chapter 10 of this volume.

13.2.3 Web Thickness

This introduces the problem of the geometry of propellant charges. The "web thickness" is a term used to describe the minimum distance through solid powder between two exposed or uninhibited surfaces. Since burning takes place on all exposed surfaces, it will be seen that the burning distance is usually one-half the web thickness. Since the rate of gas production of the propellant is proportional to the burning area (other things being equal), it is important that this area be kept constant within narrow limits, which become narrower as the pressure exponent of the powder rises. All rocket propellant charge design is based on the law of burning in parallel layers, which enables one to calculate the area of the burning surface of a grain at any time during its combustion. This law must be obeyed by any rocket propellant. Cracks, flaws, or porosity, therefore, cannot be tolerated. The problem of grain design is soluble in all cases only if powders with a wide range of linear burning rates are at hand. *The problem of propellant charge design is to arrange the geometry of the fuel in such a way that the burning surface remains essentially constant during the complete reaction, and is large enough to produce the required thrust, while the minimum distance the flame must travel is of the proper length to give the desired burning time.* This minimum distance is closely related to the "web thickness" of the powder, being equal to it or some submultiple of it.

13.2.4 Granulation

The main characteristic which differentiates rocket propellants from gun propellants is the size and shape of the individual powder grains. Gun propellant grains seldom weigh more than a few ounces, whereas rocket propellant grains may weigh upwards of 100 lb. Although they may be made in a variety of shapes, rocket propellant grains all have one characteristic in common: the shape must be such as to give approximately neutral burning. One of the chief problems in making a rocket propellant is that of granulation or forming the propellant into the desired size and shape of grain.[b]

13.2.5 Overall Specific Impulse

The specific impulse, as we have seen, is equal to the total impulse given to the rocket divided by the mass of propellant burned. A quantity of considerable use in evaluating jet motors is the "overall specific impulse" or "impulse-weight ratio" which is defined as the total impulse divided by the total weight of motor metal parts plus powder. Since the metal parts of a rocket motor are generally a dead load, the overall specific impulse is a measure of the efficiency of the design of the whole unit, and the augmentation of this quantity is an important objective in present and future rocket work. It will be recognized that at least half of the work necessary to attain this objective involves the development of lighter metal parts and is beyond the scope of this report. However, the other half presents the following problems which must be solved by the developers of propellants.

Burning at Low Pressures

The weight of the motor increases approximately in direct proportion to the internal pressure it must stand, whereas the specific impulse increases much less rapidly with pressure. There is, therefore, a distinct weight advantage to be gained by reducing the reaction pressure to a point where engineering considerations other than the bursting pressure become important factors in motor design. This requires a propellant charge whose chemical composition and geometry is such that it burns regularly at low pressures and has a low temperature coefficient.[c] A low value of the pressure exponent[d] of the propellant is advantageous on both these counts.

High Specific Impulse

It is hardly necessary to call attention to the fact that a high specific impulse of the propellant is needed to get the highest overall specific impulse of the rocket motor.

[b] This subject is covered in Section 13.3.5 and in Chapter 7.
[c] See Section 10.8.2.
[d] This exponent is n in equation (2) of Section 10.2.

DENSITY OF LOADING

It is obvious that the overall specific impulse of a given propellant and motor combination will increase as the amount of propellant per unit volume of motor increases, i.e., as the density of loading increases, and indeed will reach a maximum when the motor chamber is completely filled with powder. Limitations on the density of loading are caused primarily by the need for a large enough burning surface to produce the required thrust and by the necessity of providing sufficient port area for the gases to travel from one end of the rocket motor to another. By all odds the most effective way of obtaining a high loading density is to use a cylindrical grain which fills the motor completely and burns from one end only. This type of charge utilizes all the available space and leaves the whole cross section area available for gas flow. Its use is limited by the fact that all known propellants have too small a linear burning rate to give a large enough thrust or short enough burning time in vessels of suitable shape.

THERMAL INSULATION OF ROCKET MOTORS

The temperatures of all propellant gases are of necessity very high, and, when the burning times exceed half a second, sufficient heat is transferred to the metal parts to reduce their strength considerably. This raises the dead weight of metal needed for safe and reliable performance. Two methods of insulating the walls have been tried: the first consists of applying an insulating coating, usually a ceramic, to the interior walls of the chamber and has not been very successful; the second consists of using the propellant itself as an insulator, and this shows great promise. In such a loading arrangement the propellant is formed as a perforated thick-walled cylinder which fits tightly into the motor. The outer cylindrical surface and the fore end are treated in such a way as to inhibit burning on these surfaces. The combustion takes place in the perforation, and the hot gases impinge on only a small portion of the walls near the nozzle. Constancy of burning surface is obtained by forming the contour of the perforation into a star shape of proper size. (See Chapter 11.) This type of rocket offers the best promise for high loading density combined with light motor weight. The propellant problems presented are the granulation of powder into large perforated cylinders with thick walls and the restriction of the cylindrical surfaces.

13.2.6 Rate Control, a New Principle

Hitherto the rate of evolution of gas by a rocket propellant has been governed by the linear burning rate under the conditions in the chamber, because, by design, the burning surface itself is kept constant. In 1945 a new principle was explored by Division 8, NDRC, whereby the burning surface may actually change in area during the combustion and thereby the rate of gas evolution is made less dependent on the linear burning rate of the mass of powder. This has been accomplished by embedding in a matrix of the double-base powder strands of special powders chosen because of their low temperature coefficient and low pressure exponent. The linear burning rate of these strands determines the rate of evolution of gas by the whole mass.[e]

13.2.7 Gas Temperature

For the same expansion ratio and chamber pressure, the specific impulse of a propellant is roughly proportional to the square root of the number of moles of gas per pound and to the square root of the absolute temperature. High specific impulses are, therefore, generally accompanied by high gas temperatures. These are frequently undesirable, especially in long-burning rockets, because of the erosive effect on the nozzle. By changing the chemical composition, it is theoretically possible to produce a propellant with a high specific impulse and a fairly low gas temperature. Very little progress has been made along these lines up to date, but the problem is one of the important ones for the future.

13.2.8 Chemical Stability

A rocket propellant must conform to all the specifications required of a gun propellant in regard to stability under climatic, storage, and extreme conditions of use. The specifications, and tests to ensure conformity with them, are now well established.

[e] See Division 8 Summary Technical Report for further information on this technique.

[redacted]

13.2.9 Sensitivity

It is desirable to reduce the sensitivity of rocket propellants to impact, shock from small-arms bullets, etc., to a minimum. At present there is practically no rocket propellant which is not ignited by rifle fire.

13.2.10 Mechanical Properties

The propellant in a jet-operated motor is subject to a variety of stresses during its use. These stresses come from differential gas pressure in the motor itself and from setback forces arising from acceleration or from shock during handling. Rates of applications of these stresses are, in general, quite high, and it is essential that measurements made in the laboratory to test the physical properties of rocket propellants should be made with comparable loading schedules. Although a considerable amount of work has been done on the measurement of physical properties of rocket propellants, it has not yet been established what are the really significant measurements to be made. It seems, however, that Young's modulus, the impact resistance, plastic flow, and failures in tension and compression all give results of practical significance if measured over an appropriate range of loading rates.

13.3 DOUBLE-BASE POWDERS

13.3.1 General Description

The name "double-base powder" was originally given to colloidal propellants containing two bases or materials capable of self-combustion, namely, nitrocellulose and nitroglycerin. It has been extended to include all propellants made with nitrocellulose and one or more explosive plasticizers such as nitroglycerin, diethylene glycol dinitrate, and DINA.[f] In addition to nitrocellulose and the explosive plasticizer, these propellants usually contain a stabilizer such as centralite and auxiliary plasticizers such as centralite, phthalate esters, triacetin, dinitrotoluene, and other compounds of this nature which also act as cooling agents. In order to suppress flash and to obtain smoothness of burning at low temperatures, it has been found desirable to add 1 or 2 per cent of a potassium salt to double-base powders. By adjusting the amounts of nitrocellulose, the physical properties of the colloid may be varied over a wide range of toughness and plasticity, and by varying the amount of explosive plasticizer and coolants the flame temperature and the burning rate may also be given wide variations. Several double-base compositions are shown in Tables 1, 2, and 3.

13.3.2 Thermodynamic Properties

The densities of most double-base powders are approximately 1.6 grams per cu cm, that is, about 0.058 lb per cu in. The isobaric adiabatic flame temperatures vary from 2400 to 3200 K. The specific impulses vary from 235 lb-sec per lb for the powders containing approximately 40 per cent nitroglycerin and 2 or 3 per cent of cooling agent, to 190 for powders containing 20 per cent nitroglycerin and approximately 20 per cent of cooling agent. The number of moles of gas per gram is about 0.040.

13.3.3 Burning Properties

At room temperature (70 F) the linear rates of burning of double-base powders vary between 0.4 and 1.2 ips at 2,000-psi pressure. These figures correspond to rates of gas evolution of 0.024 and 0.071 lb-sec per sq in. of burning surface under these conditions. It is a general rule that the hotter the powders, i.e., the higher the adiabatic flame temperature, the higher the burning rates. It is of interest to note that at 2,000-psi chamber pressure 1 sq in. of burning surface gives a thrust of 4.5 lb force with the cooler powder and 16.3 lb force with the hotter in motors of appropriate design.

Until recently the pressure exponents (see Chapter 9) of all known double-base powders were undesirably high, being between 0.7 and 0.8. This caused irregularity of burning, forced a reduction in loading density, and accentuated the temperature coefficient of the chamber pressure and thrust. Indeed for a fixed rocket geometry the pressure and thrust increased approximately 0.8 per cent per degree Fahrenheit. Rather severe practical limitations to the use of double-base powder rockets arose from this. Indeed for a long time double-base powder was ruled out from con-

[f] Diethanolnitramine dinitrate.

sideration in connection with JATO units on this account. Recently several double-base compositions have been discovered whose pressure exponents in the range 800 to 2,000 psi are 0.5 or less. When these propellants are used, the pressure and thrust of a rocket motor change only 0.2 to 0.3 per cent (and even as low as 0.05) per degree Fahrenheit over the temperature range −40 to 140 F. One of these powders is particularly adaptable to use at 1,000-psi pressure. Unfortunately all are cool powders and have relatively low burning rates. The reason for the low pressure exponent of these double-base powders is not yet completely understood, but 1945 experiments on a captured Japanese powder give a clue which should certainly be followed.

13.3.4 Mechanical Properties

If properly made, double-base powders can be obtained as tough, nonporous, homogeneous colloids which obey perfectly the law of burning in parallel layers. The mechanical strength and elastic properties such as Young's modulus rise rapidly with the nitrocellulose content. It should be noted that double-base powders colloided with the aid of an active solvent are much stronger and tougher than those made by rolling and dry extrusion. In general, the mechanical and elastic properties of the better developed double-base powders are adequate at 70 F to stand the stress set up during the projection of a rocket. At high temperature, i.e., above 100 F, experience has shown that these propellants flow too easily and have too low a value of Young's modulus to be satisfactory. Furthermore, at low temperature their impact strength falls off so rapidly that powder breakup from brittle fracture occurs during the launching of many rockets. These defects have been studied, but, although promising clues have been found, a considerable amount of research work is necessary to put this aspect of the subject on a sound theoretical and practical basis.

13.3.5 Granulation

Double-base powder may be made in grains suitable for use in rockets by four different processes, each of which has its own advantages and limitations. These are (1) solvent extrusion, (2) solventless extrusion, (3) casting, (4) pressure molding.

Solvent Extrusion

In this process an active volatile solvent is added to the nitrocellulose-nitroglycerin mixture, and the whole is stirred in an incorporator. The solvent swells the nitrocellulose and permits colloiding, i.e., breakdown of the fibrous structure, with a small amount of mechanical work. The soft paste or

Table 1. Nominal compositions of standard double-base powders.

Ingredient \ Powder	JPT	JPT M13	T–2 (H–4)	⅜-in. Stick	JPN	Cordite S.C.	Cordite SU/K	Cordite R.S.
Nitrocellose	58.80	57.30	58.00	58.25	51.50	50.00	50.00	57.00
Per cent nitration	13.25	13.25	13.15	13.25	13.25	12.20	12.20	12.20
Source*	WP or CL	WP or CL	WP or CL	WP or CL	CL	WP	WP	WP
Nitroglycerin	40.00	40.00	30.00	41.00	43.00	41.00	41.00	28.00
2–4 Dinitrotoluene			2.5					11.00
Ethyl centralite	1.00	1.00 to 3.00	8.00		1.00	9.00	9.00	4.00
Diphenylamine	0.2			0.75				
Diethylphthalate					3.25			
Potassium sulfate		1.50	1.5		1.25†			
Potassium cryolite (added)							2.25	
Carbon black (added)		0.05	0.02		0.2			
Methyl cellulose (added)					0.1			
Candelilla wax (added)					0.075			
Lead stearate (added)		0.015						
Heat of explosion (water liquid basis) cal per gram	1300		930	1316	1230	960	955	900

* *WP* = wood pulp.
CL = cotton linters.

† Not included in heat of explosion calculations.

dough so formed is extruded through dies of the proper size and shape and cut to length. The solvent is removed by drying at elevated temperatures in forced-air-dry houses. Powder granulated by this method is tougher and harder than powder of the same composition granulated by other methods. It is, therefore, indicated in cases where high accelerations are needed, because its fibrous structure helps resist fracture by the setback forces. The action of the solvent in reducing the explosive power and sensitivity of the paste reduces hazards of manufacture. The chief disadvantage of solvent-extruded powder is the severe limitation on the web thickness imposed by the necessity of removing the solvent. It is not feasible to produce this powder with web thicker than half an inch because of the very long drying time and the production of cracks during shrinkage, attendant on the solvent removal. Furthermore, exact control of shape and dimensions in very difficult in solvent extrusion.

TABLE 2. Nominal compositions of promising experimental double-base powders.

Ingredient \ Powder	H-5	L 4.8	G 117 B	JPH
Nitrocellulose	58.00	58.50	50.00	54.50
Per cent nitration	13.25	13.20	13.25	12.60
Source	WP	WP	WP	CL
Nitroglycerin	20.00	22.50	30.00	43.00
Dinitrotoluene	2.50	2.50	14.50	
Ethyl centralite	8.00	8.00	4.00	1.00
Triacetin	10.00	8.50		
Potassium sulfate	1.50		1.50	1.50
Carbon black (added)			0.02	0.10
Lead stearate (added)	0.40	0.40	0.40	
Heat of explosion (water liquid cal per gram)	632	699	940	1252

SOLVENTLESS EXTRUSION[g]

In this process the nitrocellulose-nitroglycerin mixture is colloided by severe mechanical working on heated rolls without the action of a solvent. The resulting sheet powder is extruded hot (110 to 170 F) through appropriate dies, annealed, and is then ready for use. In this process the web thickness is limited only by the sizes of press available. At present, with the 18-in. press at Inyokern, powder grains with cross section areas equivalent to a circle 9 in. in diameter can be successfully extruded with webs 3 in. or larger. Exact control of shape and size is readily feasible in solventless extrusion, but the powder extruded by the solventless process is not as tough and strong as solvent powder. The process is quite hazardous, heavy and costly machinery and barricades being required. It is, however, the most important source of rocket propellants now available. Examination of German and Japanese propellants indicates that they are definitely stronger than those produced in this country and presents a clue to the improvement of the strength of solventless double-base powder that should be followed at once.

[g] See Chapter 7.

TABLE 3. Nominal composition of cast double-base propellant.

Only one powder has been investigated thoroughly enough to warrant it being considered for standardization. Its composition is given as follows:

Matrix

Casting powder	35 parts by volume
Casting solvent	12 parts by volume

Casting powder—granulated in cylinders 0.030 in. diameter, 0.030 in. long.

Nitrocellulose (13.15 per cent nitrogen)	74.0
Nitroglycerin	20.0
Diethylphthalate	5.0
Ethyl centralite	1.0
Carbon black (added)	0.5

Casting solvent

Nitroglycerin	64.0
Dimethylphthalate	35.0
Ethyl centralite	1.0

Rate control strands

Nitrocellulose (12.6 per cent nitrogen)	25.0
Potassium perchlorate (3 microns)	56.0
Carbon black	9.0
Ethyl centralite	1.0
Plasticizer	9.0

The plasticizer consists of 74 per cent nitroglycerin, 25 per cent dimethylphthalate, and 1 per cent centralite. 0.28 per cent magnesium stearate is added to the whole mixture.

CASTING

The starting material for this process is finely granulated, previously colloided powder, such as double-base rifle or pistol powder. Cut or ball powders are both serviceable. The small particles of this nitrocellulose-nitroglycerin powder are mixed with a sufficient quantity of an active, nonvolatile, casting solvent (e.g., nitroglycerin dissolved in triacetin) to form a pourable slurry. This is cast into a mold which may be a metal container or a plastic tube. The latter may serve as a restricting material if this is desired. Heating for about a day at 60 C causes the mass to set to a tough nonporous grain which has entirely satisfactory burning properties. Provided that care is taken in the selection of the composition, there seem to be no limits to the size and shape of grains that may be produced by this process, and it is particularly well suited to the production of large single grain charges. The casting process is also well adapted for applying the principle of burning rate control by strands of special powder, since the propellant may be cast directly around the strands. Cast propellants are still in the development stage, but they offer many advantages. In addition to those already given, the simplicity of the equipment and the cheapness and comparative safety of the process may be cited.

By July 1947, Allegany Ballistics Laboratory, operated by the Hercules Powder Company, had carried the development of one type of cast double-base propellant to the stage where rocket thrust units carrying single grain charges weighing more than 600 lb were fired successfully in flight under conditions of extreme acceleration.

Pressure Molding

Molded double-base powder has been produced by mixing Western Cartridge "Ball Powder" with a few per cent of plasticizer and molding it into a large grain by the application of heat and pressure. The details of the process have not been published, and indeed the whole work is in a fairly elementary stage. Its significance has been greatly diminished since the development of the casting process.

Summary

The granulation processes just described cover the field of rocket propellants very adequately. The solvent process is useful where thin-web grains strong enough for rapidly accelerated rockets are desired. The solventless process is well adapted to produce large grains whose lengths are large compared with their diameters—there is, however, an upper limit to the diameter. The casting process is best adapted to producing grains whose lengths and diameters are comparable. It is especially suited to the fabrication of large-diameter grains. The larger the grain, the more economical is the casting process.

13.4 CAST PERCHLORATE PROPELLANTS

13.4.1 General Description

These propellants are made by mixing intimately together finely powdered potassium perchlorate and an organic binder in a fluid condition. The mix is cast into a mold where it solidifies by thermoplastic or thermosetting action. More recently ammonium perchlorate has been used instead of potassium perchlorate to cut down the smoke. As organic binders asphalt and oil mixtures, paraplex styrene resins, rubber bases, and fusible ethylcellulose-castor oil mixes have been used with success. The great advantage of these propellants is the extraordinarily simple process by which they are produced and the cheapness of the materials involved. It may be noted as a matter of interest

Table 4. Nominal compositions of some cast perchlorate propellants.

Ingredient \ Powder	ALT–39 (Aerojet)	Galcit 61–C (Aerojet)	MA–70	MA–142	Bruceton cast perchlorate
Potassium perchlorate	75.0	75.5	....	....	74.5
Ammonium perchlorate	....	....	75.0	74.75	....
Base	25.0*	24.5†	25.0‡	25.0§	25.0‖
Catalyst (chromic oxide)	....	....	....	0.25	....
Carbon black	....	....	....	....	0.5

* Asphalt, Union LT–1 (AMS–C15) 90 per cent, oil (AMS–C3) 10 per cent.

† Asphalt (AMS–C2) 70 per cent, oil (AMS–C3) 30 per cent.

‡ Asphalt, LT–1 (AMS–C15) 42 per cent, paraplex RG–2 38 per cent, dibutyl sebacate 8 per cent, Acrawax C 12 per cent.

§ Asphalt, LT–1 (AMS–C15) 34 per cent, paraplex RG–2 46 per cent, dibutyl sebacate 8 per cent, Acrawax C 12 per cent.

‖ Permafil 2851 98.2 per cent, tertiary butyl perbenzoate 1.3 per cent, lecithin 0.5 per cent, quinone 0.03 per cent.

[redacted]

that these are the only solid propellants that do not depend ultimately on nitric acid. It is impossible to give in detail the properties of all propellants of this type that have been studied, so that attention will be focused on three propellants, one of the asphalt and potassium perchlorate type, one of the asphalt-ammonium perchlorate type, and one of the ethylcellulose-potassium perchlorate type. Compositions are shown in Table 4.

13.4.2 Asphalt-Potassium Perchlorate Propellant—Galcit 61-C

This propellant is made by stirring together finely ground potassium perchlorate and a hot asphalt-oil mixture, pouring into the motor, which has been lined with a layer of asphalt, and allowing to cool. Alternatively, it may be cast into a mold, removed, and coated with asphalt and tape or some other restricting medium.

Thermodynamic Properties

The density of this propellant is high, being 1.75 to 1.82 g per cu cm, i.e., 0.063 to 0.066 lb per cu in. The adiabatic flame temperature is calculated to be 2100 K. There are uncertainties in this calculation, and this figure is probably too low. The gases from this propellant erode the nozzle severely. The specific impulse with chamber pressure of 2,000 psi and reasonable expansion ratio is 170 to 180 lb (force) $\times$ seconds per lb. The number of moles of gas per gram is 0.036. The ratio of the specific heats at constant pressure and constant volume is 1.21. The propellants yield a dense white smoke on burning. Galcit 61-C, like others of this type, is very stable and difficult to ignite.

Burning Properties

At 60 F the linear burning rate is 1.5 $\pm$ 0.1 ips at 2,000-psi chamber pressure; this corresponds to a gas production of approximately 0.098 lb per sec per sq in. of burning surface. The corresponding value of the thrust developed by the burning of 1 sq in. of surface is 17 lb (force). The pressure exponent of this powder has not been well investigated, but it is undesirably high, being about 0.75. On the other hand, the temperature coefficient of the isobaric burning rate is so low that the variation of thrust with temperature in a given rocket is only about 0.35 per cent per degree Fahrenheit under conditions of use.

Mechanical Properties

Over the usable temperature range these powders are fairly soft and not brittle enough even at low temperatures to be easily fractured by rough handling, although cracking from thermal stresses at low temperatures has been troublesome. When directly supported by the motor walls, the propellant has adequate strength to withstand the stresses encountered in service, but it seems certain that over most of the service temperature range the propellant is too soft for applications such as radial burning where it is supported only at one end. The mechanical properties of this propellant determine the safe operating temperature limits. At high temperatures the material becomes soft enough to flow, whereas at low temperatures it hardens to a point where shrinkage cracks appear. The improvement of the physical properties of this propellant has been a problem of urgency and led to the development of the ethylcellulose and paraplex-binders.

13.4.3 Asphalt-Ammonium Perchlorate Propellants

A number of these propellants have been developed with a view to eliminating or cutting down the amount of smoke produced by cast potassium perchlorate mixtures. These are made in essentially the same manner as the asphalt-potassium perchlorate propellants. They contain in addition to ammonium perchlorate and asphalt small amounts of other plastics and plasticizers, together with chromium trioxide which acts as a catalyst.

Thermodynamic Properties

The densities of propellants of this type range from 1.52 to 1.56 g per cu cm, and specific impulses varying from 150 to 190 are reported. The gas contains 0.050 moles per g, and the adiabatic flame temperature is given as 1830 K. There is considerable uncertainty in these figures.

Burning Properties

At room temperature the linear burning rates of the ammonium perchlorate propellants are much lower than those of the potassium perchlorate propellants, varying from 0.4 to 0.85 ips at 2,000-psi pressure. Some compositions have been made which burn well at 1,000 psi. With these propellants, burning at 2,000-psi pressure, thrusts varying between 4 and 8.5 lb (force) per sq in. of burning surface may be obtained. The pressure exponents and the temperature coefficients have not been investigated.

MECHANICAL PROPERTIES

The mechanical properties are quite similar to the asphalt-potassium perchlorate propellants which have already been described.

13.4.4 Ethylcellulose-Potassium Perchlorate Propellants

These propellants are in the advanced experimental stage but could be developed and applied fairly readily. One type is made by mixing a hot molten ethylcellulose-castor oil mixture with potassium perchlorate and aluminum or carbon and casting the mix into a suitable mold or vessel. On cooling, the mass sets up to a tough solid, which has better mechanical properties over a wide temperature range than does the asphalt composition. Another type is made by mixing the perchlorate and aluminum or carbon with the General Electric Company's resin "Permafil," which can be cast at room temperature and hardens without shrinkage to a rubbery solid of unlimited temperature range.

These propellants have thermodynamic and burning properties very similar to the asphalt-potassium perchlorate ones, but with the significant difference that the added aluminum or carbon brings the pressure exponent down from 0.7 or 0.8 to 0.6, a very important reduction. Further investigations of the effects of substances like aluminum on the pressure exponent are strongly indicated as a means of improving this type of propellant.

13.5 MOLDED COMPOSITE PROPELLANT

13.5.1 General Description

These propellants are prepared by milling together in edge-runner mills a mixture of ammonium picrate, alkali nitrate, and a small portion of a resinous binder. The powdery product from the mills is forced into grains of the desired size and shape by compression molding at about 10,000 psi in a large hydraulic press. The grains are cured at a predetermined temperature for a fixed time before use. Because of the nature of the binder used, these grains can be easily restricted by a plastic coating, which prevents burning on the inhibited surfaces.

The fabrication of this propellant requires a large number of small edge-runner mills, although improved techniques may probably be developed by further investigation. It also requires large presses, and a considerable number of these are necessary because of the comparative slowness of the molding operation. The raw materials for this propellant are all currently manufactured in large amounts. Molded composite propellants produce considerable amounts of white smoke, the quantity being smaller with the slower burning compositions which contain smaller proportions of alkali nitrate. Although a large number of these composite propellants have been studied, it has been found possible to cover a wide range of properties with four compositions: CP 401, CP 404, CP 218B, and CP 492. These are arranged in order of increasing burning rate. Their nominal compositions are shown in Table 5.

TABLE 5. Nominal compositions of certain molded composite propellants (OSRD Report No. 5700).

Ingredient \ Powder	CP 401	CP 404	CP 218B	CP 492
Ammonium picrate	72.0	54.0	46.5	41.0
Sodium nitrate	....	....	46.5	....
Potassium nitrate	18.0	36.0	....	50.0
Plastic binder	10.0*	10.0*	7.0†	9.0‡
Zinc stearate (added)	0.4	0.4	0.4	0.4

* 5.0 per cent ethylcellulose, 5.0 per cent Aroclor No. 1254.
† 5 per cent buramine resin, 1.5 per cent Santicizer No. 8, 0.5 per cent butanol.
‡ 4.5 per cent ethylcellulose, 4.5 per cent Aroclor No. 1254.

13.5.2 Thermodynamic Properties

The densities of molded composite propellants range from 1.66 to 1.79 g per cu cm or 0.060 to 0.065 lb per cu in. Reliable estimates of the flame temperatures have not been made. The specific impulses, measured at chamber pressures in the vicinity of 1,000 psi and with the optimum expansion ratio, lie between 160 and 170 and vary little with the composition of the propellant.

[redacted]

13.5.3 Burning Properties

All molded composite propellants burn very well at low pressures; indeed 500 to 1,000 psi seems to be the optimum chamber pressure for these fuels. By change of composition a wide range of burning rates may be obtained without much change in specific impulse. For example, at 1,000-psi pressure and 70 F the linear burning rates of CP 401 and 492 are 0.24 and 1.0 ips respectively. The corresponding gas production figures are 0.014 and 0.064 lb per sec per square inch of burning surface, and this gives thrusts per square inch of burning surface of 2.3 and 10.6 lb (force). It should be noted that the ratio of nitrate to picrate is the principal factor in determining the linear burning rate, but the particle size of the nitrate is also an important factor in those powders which contain potassium nitrate. The pressure exponent in the burning rate law for all these composite propellants is quite low, being on an average 0.5. This promotes stability of burning at high loading densities and gives a very small effect of temperature on the pressure and thrust of a given motor. Indeed the pressure in a motor charged with a molded composite propellant changes only 0.22 per cent per degree Fahrenheit.

13.5.4 Mechanical Properties

When properly made, molded composite propellants are nonporous solids with a smooth hard surface. They obey the law of burning in parallel layers. It is essential that the density be controlled in manufacture so that it is between 0.950 and 0.965 times the theoretical fully packed density. If the density is below this limit, troubles from porosity will arise, whereas, if it exceeds this limit, the grains may crack on being removed from the mold.

13.5.5 Compression Strength

All the composite propellants will withstand compressive stresses of 3,000 psi for short times even at 60 C and several times this amount at room temperature. Since these materials are plastics, the value of the compression strength depends on the rate of loading, and few laboratory measurements under these conditions have been made. However, numerous tests of propellant grains in rockets subjected to excessive acceleration have failed to give any evidence of compression failures.

13.5.6 Impact Resistance

Molded composite propellants have a very low impact resistance; it is about one-tenth that of double-base propellants. However, simple shock-absorbing mountings made from cork have been devised [h] which enable the propellant grains to stand up against any rough usage tests, such as dropping on concrete, which do not damage seriously the metal parts of the rocket motor.

13.5.7 Thermal Shock

The resistance to thermal shock leaves something to be desired. It depends on the size of grains and the severity of the temperature change, and is in the state where it is quite advisable to examine the effects of thermal shock on any new rocket loaded with a molded composite propellant. The chemical, thermal, and explosive stability of all composite propellants of this type is extremely high, and the impact sensitivity is low.

13.5.8 Granulations

The pressure-molding process works best when the diameter of the grains is approximately equal to the length. Grains whose lengths are much greater than their diameters must be produced by the cementing together of one or more smaller grains. Since adequate cements are available this condition introduces no great difficulties and makes possible the production of a wide variety of shapes and sizes. Up to 1946, grains varying from 1 to 12 in. in diameter and from 1 to 51 in. in length had been successfully made. The only limit to the diameter is the size of the press available.

13.6 SOLVENT-EXTRUDED COMPOSITE PROPELLANTS

These propellants consist of a filler composed of carbon black and either potassium perchlorate or potassium nitrate dispersed in a binder of double-

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[h] See Division 8 Summary Technical Report.

[redacted]

base powder. The proportions are usually 65 per cent filler, 35 per cent binder, although for some purposes where reduction of smoke is important the fraction of filler has been reduced to 9 per cent.

In addition to apparatus for grinding the perchlorate or nitrate the equipment needed for making these powders is the same as for making solvent-extruded double-base powders, and the same limitations of web thickness apply.

The great advantage of solvent-extruded composite propellants lies in the small value of their pressure exponent which is approximately 0.45 and which permits high loading density and cuts down the temperature coefficient of pressure and thrust. The specific impulses are about the same as those of double-base powders, and the compositions may be adjusted to cover a wide range of burning rates. Indeed, extruded composite propellants with burning rates faster than are feasible with double-base powders are readily obtainable. The granulation limitations described under solvent-extruded double-base powder apply to solvent-extruded composite propellants, and their main use is limited to relatively fast-burning rockets or to the rate control strands which are used in conjunction with cast double-base powder grains. For this latter purpose, strands of composite propellant are ideal because of the small effects of pressure and temperature on their burning rates, and because of the wide range of burning rates that may be realized within the composition scope of solvent-extruded composite strands.

13.7 PLASTIC PROPELLANTS

This type of rocket propellant has been developed by the British and is quite similar in composition and ballistic properties to the American molded composite propellants. The main difference lies in the binder, which is more fluid and present in larger amounts, so that the plastic propellant does not set up to a hard mass but retains a puttylike consistency. It is molded directly into the rocket motors under fairly low pressure in the form of central-burning charges inhibited on the outer surface by the motor walls, a reliable bond between the plastic propellant and the steel wall having been developed. The puttylike consistency of the propellant allows it to expand or contract with the motor wall without the setting up of stresses large enough to cause rupture or cracking.

TABLE 6. Nominal compositions of some solvent-extruded composite propellants.

Ingredient \ Powder	EJA	EJB	MJA	T-4
Nitrocellulose	21.00	42.00	26.00	54.60
Per cent nitration	12.66	13.10	13.10	13.15
Nitroglycerin	13.00	26.50	21.50	35.50
Ethyl centralite	1.00	....	2.50	0.9
Potassium perchlorate	55.50	25.50	....	7.80
Potassium nitrate	....	....	43.00	....
Carbon black	9.00	4.20	7.00	1.20

The thermodynamics and burning properties of this propellant are similar to those of the slower burning molded composite propellants.

13.8 RECOMMENDATIONS FOR FUTURE WORK[1]

The foregoing review of the status of solid rocket propellants suggests strongly certain lines along which future work should proceed and makes possible several general recommendations which will be advanced in the following. It should be noted that these recommendations are of a general nature and are independent of any programs that might already be planned for the development of specific devices. Future work to be undertaken falls naturally into two classes:

1. *Development work*, which includes the improvement of existing propellants and especially the development to an entirely satisfactory state of a few solid propellants which cover the range of foreseeable requirements. The scale of this type of work is on a pilot plant or higher level, and its main object is to render available to the United States reasonably satisfactory propellants which may be prepared in quantities at short notice.

2. *Research work*. This includes work on a laboratory level that is designed (a) to make radical improvements in existing types of propellants, and (b) to broaden the whole basis underlying the art of propellant manufacture.

The discussion in this section will be classified according to the types of propellants considered.

[1] These recommendations were made early in 1946. Many of them were put into effect during preparation of this volume.

[redacted]

13.8.1 Solvent-Extruded Double-Base Powders

It is recommended that no further development work be conducted on powders of this type since three satisfactory powders are now available; namely, T–1 powder, the T–4 (BBP), and the T–2 powder. Specifications for these have been written, and the only problems that require consideration are those dealing with improvement of the control in large-scale production.

13.8.2 Solventless Double-Base Powders

Compositions which cover the whole range of calorific values or burning rates have been investigated for this type of powder. It is recommended that work leading to the development of three compositions which are satisfactory from the ballistic and manufacturing points of view be undertaken at once. It is further recommended that these powders be based on JPN, the high-calorific powder, G 117B, medium-calorific powder, and L 4.8 (note: the compositions of these powders as known at present are given in Tables 1 and 2). These powders cover the range of burning rates obtainable with double-base propellants, and the problems connected with their manufacture are known to be soluble. Considerable improvement in the manufacturability should be sought, but no sacrifice in ballistic qualities such as smoothness of burning and small temperature coefficient should be made. It is emphasized that these problems are fairly short range in nature, but they should not be regarded as solved until the results have been tested on a large scale, since quantity production is an important object.

It is also recommended that immediate steps be taken to use existing lines of evidence to improve the mechanical properties of these powders, particularly the resistance to load at high temperatures and the "brittleness" at low temperatures.

It is also recommended that studies be made of the effect of newly developed stabilizers in extending the safe life and cutting down the gas production in double-base powders. The gas production in these powders is now thought to be the major cause of cracking during high-temperature storage, an effect which at present imposes serious limitations in the use of double-base powder in large web grains.

Problems concerned with the extrusion of solventless powders in very large grains, for example, 6 to 10 in. in diameter, should receive high priority. This is particularly true for grains restricted on the outer surface and having a star-shaped perforation, since this type of grain gives the highest promise of realizing the largest overall specific impulse in rocket motors. The need for motors with high impulse and low weight for the launching of guided missiles and similar devices becomes more urgent every day.

13.8.3 Cast Double-Base Propellants

This development, particularly with the use of rate control strands, is regarded as one of the most promising in the whole field, and it should be pursued vigorously, particularly in view of the increasing demand for jet-operated thrust units of larger and larger size. It is suggested that attention be given to the development of approximately three compositions or combinations of compositions in burning rate and rate control strands covering the same range as that indicated in the solventless-extruded powder field. Attention should also be given to the development of large radial-burning grains with a low temperature coefficient, produced either by adjustments of the composition of the powder or by the use of rate control devices. The preparation of cast double-base grains in very large sizes or with star-shaped perforations should also receive early attention. In this connection the development of a smokeless composition with the mechanical properties and adhesive qualities of the British plastic propellant would fill a pressing need.

13.8.4 Pressure Molding of Double-Base Powder

It is recommended that very low priority be given to this type of development in the future, since the casting process is simpler and leads to the same results. Furthermore, the experience of the years 1942-45 does not justify optimism concerning further work on pressure molding of double-base powder.

13.8.5 Cast Perchlorate Propellants

The cheapness and availability of these propellants and the possibility of obtaining high loading densities suggest very strongly that development work to improve them should be pursued vigorously. Careful attention should be paid to the chemical engineering problems arising in the manufacture so that a more uniform product may be obtained. Higher mechanical strength and a wider usable temperature range are important objectives that should be sought. Improvement of the exponent in the burning law is also a very necessary development; clues to this already exist in the action of aluminum in some of these propellants. It is also recommended that attempts be made to increase the burning rate of the smokeless ammonium perchlorate propellants or to develop other cast compositions with high burning rate and low smoke.

13.8.6 Molded Composite Propellants

These are in a fairly well-developed state, the only problems really requiring attention being improvement of the manufacturing process, better control of the uniformity of the product, and removal of any instability at high chamber pressure. It is felt that the field covered by molded composite propellants can, in general, be covered by others that are more promising in their properties or easier to make. Hence it is not recommended that an extensive development program be conducted on this work. Since molded composite propellants are the only ones now available for large thrust units designed to give very high thrusts, it is strongly recommended that facilities for making this propellant be kept in working order until a completely satisfactory replacement has been developed.

13.8.7 Solvent-Extruded Composite Propellants

At present these propellants with their low exponent in the burning law and high rate of burning are the best known for rate control applications. The development problems of the solvent-extruded composite propellants consist mainly in the securing of positive manufacturing control and should be pushed to a point where satisfactory specifications for manufacture and quality may be written.

13.8.8 Plastic Propellants

These are receiving attention in Great Britain, and, since they have the ballistic properties of the molded composite propellants and there is a possibility that the same mechanical properties may be developed in cast double-base propellants, it is recommended that little work along this line be done until the possibilities of plastic cast double-base propellants are more thoroughly explored.

13.9 RECOMMENDATIONS FOR RESEARCH WORK

The research program recommended recognizes two main objectives:

1. Radical modifications and improvements of existing propellants; for example, replacement of nitroglycerin by a new explosive plasticizer.
2. The production of entirely new types of propellants with different bases; for example, use of high polymers other than nitrocellulose.

These new propellants will, of course, recommend themselves because of outstandingly good physical or burning characteristics, or great ease or flexibility of manufacture. Such a research program must be guided by an understanding of the fundamental characteristics involved; namely, the mechanism of burning, the control of the physical and mechanical properties, and the knowledge of the desirable properties of new ingredients and methods of making them.

The research program suggested here, therefore, falls into three classes, which not only subdivide the research problems into natural groups, but also indicate an organization for carrying them out. In the following pages this program is set out, first in summary and secondly in more detail, so that the reader may comprehend its scope more readily.

13.10 SUMMARY OF RESEARCH PROBLEMS IN THE FIELD OF ROCKET PROPELLANTS

13.10.1 Class A. Theory of the Burning of Rocket Propellants (Kinetics of Powder Reactions)

The objective of this class of problem is an understanding of the relations between those quantities of significance in the burning of rocket propellants and those quantities which may be controlled in their manufacture. A satisfactory theory should enable one to predict the burning properties of a propellant from its composition and to make powders with burning properties specially adapted to certain purposes. A sound theoretical basis is of utmost importance in the guiding of work in the whole program.

13.10.2 Class B. Physical Theory of the State of Colloidal or Other Solid Propellants (Statistical Mechanics of Solid Propellants)

This general class of problems is concerned with the relation between the molecular properties of the ingredients (chemical nature, degree of polymerization) and the physical and mechanical properties of the resultant mass. Practical questions, such as extrudability, strength, "degree of colloiding," control of soundness and to some extent of burning properties, fall into this class. It is highly probable that organic high polymers of one sort or another will continue to form the basis of solid propellants for some time to come. The search for a complete understanding of the relationship between the characteristics of the molecules and the properties of the solid or liquid state in highly polymerized systems is one of the most vital physicochemical problems of the day, and one which links up the study of propellant explosives with that of other plastics.

13.10.3 Class C. Fundamental Developments of New Propellants (Chemistry of Propellants)

The work covered by this class depends for its success on close coordination with the work listed under classes A and B, because these classes cover fields nearer to the ultimate application. Class C, however, is sufficiently varied and specialized to merit separate consideration. In this class is considered not only the chemistry of old or entirely new powder ingredients, but also the search for new ways of restricting the burning of solid propellants, the development of semisolid propellants, the development of fuels with low flame temperatures but high specific impulses, and the development of powders with improved ignitibility.

13.10.4 More Detailed Outline of Program

CLASS A. KINETICS OF PROPELLANTS

1. Theory of burning of solid propellants.
   a. Study of reaction in solid.
   b. Study of reaction in gas phase.
   c. Influence of environment, pressure, temperature, radiation on kinetics (rate and exponents, etc.).
2. Thermodynamic studies of powder and powder gases.
   a. Specific heat measurements.
   b. Heats of reaction.
   c. Thermal conductivity of propellants.
   d. Temperature measurements near reaction zone.
3. Isolation and identification of intermediate decomposition products.
4. Laboratory studies of kinetics of intermediate reactions, i.e., reactions in which the known intermediate products take part.
5. Application of theory to specify desirable powder ingredients and tests of predictions.
6. Effect of powder composition and burning properties. Under this subhead we include all studies of types A1 to A4 as applied to nitrocellulose powders, composite propellants, and powders with entirely new bases and plasticizers.
7. Development of new techniques for studying the kinetics of reaction of gaseous, liquid, and solid propellants, for example, application of mass spectrograph, radioactive tracers, and high-speed photography.

CLASS B. PHYSICAL STATE OF PROPELLANTS

1. Systematic studies of physical properties of existing and new propellants over range of pressure, temperature, and rate of application of stress.
2. Systematic study of plastic properties over ranges of pressure and temperature.

[redacted]

3. Studies of molecular characteristics and fundamental chemistry of cellulose, nitrocellulose, and other high polymers. Structure of solid propellants. Relation of molecular characteristics, e.g., molecular weights and polar groups, to the properties of the solid, e.g., degree of colloiding of powder.

4. Effect of mechanical working and other external effects on molecular characteristics and structure of propellants.

5. Study of molecular interaction of plasticizers with propellant bases. Influence of bonding on physical state of the solid propellants.

6. Fundamental studies of the adhesion of solid propellants to metals, plasticizers, etc.

7. Development of new apparatus and techniques for studying the molecular properties and the macroscopic structure of solid fuels.

Class C. Chemistry of Propellants

1. Synthetic organic chemistry as applied to explosive bases.

2. Synthetic organic chemistry as applied to explosive and nonexplosive plasticizers.

3. Use of new bases and plasticizers to obtain propellants with higher specific impulses but low flame temperatures.

4. Studies of new stabilizers and their action. Reduction of gas formation. Improvement of high-temperature properties.

5. Use of ingredients to promote ignitibility of powders.

6. Exploration of new manufacturing methods, including entirely new colloiding processes.

7. Development of semisolid propellants—thickened monofluids.

8. Development of restrictive coatings and methods of application.

9. Investigation of thermodynamic and thermal properties of powders and powder constituents. This is particularly important in the case of new constituents.

10. Application of new methods to chemical and physical analyses of powders.

13.10.5 General

The program just outlined is given in fairly general terms, but it covers the avenues of investigation that now[j] seem worth following and provide fairly well-defined objectives. The details should, of course, be filled in more completely by those who are to undertake the work. It is suggested very strongly that one of the first steps to be taken by those undertaking the job should be the preparation of a monograph giving the present status of solid rocket propellants. In this way the outstanding problems will be brought into sharp relief.

[j] In early 1946.

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PART IV

ROCKET WEAPONS AS DEVELOPED AND USED IN WORLD WAR II

By *C. W. Snyder* [a]

A SIGNIFICANT DEVELOPMENT of World War II was the resurgence of the artillery rocket as a major weapon. This is strikingly illustrated by the fact that in 1941 the U. S. Navy had no rocket weapons and evinced little interest in them, whereas in 1945 the Navy was spending on them $100,000,000 a month—more than on all its other types of ammunition combined.

All the Navy's rocket weapons, as well as a considerable portion of those used by the Army, were developed by OSRD's rocket project at the California Institute of Technology, Contract OEMsr-418. The CIT work began in September 1941, expanded rapidly, and continued intensively into late 1945. Many reports were issued during this period. Two monographs and seven final report volumes on rockets, prepared under the contract, recapitulate the principal results and conclusions of four years of high-pressure activity.[b]

The following chapters attempt to provide an introduction to these volumes, and to summarize them in part, primarily for the benefit of those who may be concerned with rocket research in the future.

Since one of the major aims is to explain why CIT rockets evolved as they did, certain basic factors are given here in the beginning. They are

1. *Propellant.* The only rocket propellant which could be made available in sufficient quantities to meet the requirements of an artillery weapon was ballistite. It was far from ideal for the purpose.

2. *Simplicity.* The keynote of all designs was simplicity. From the beginning the group set for itself the task of developing to the utmost the simplest kinds of rockets, which could be made in enormous quantities cheaply and quickly, in the belief that this course was more likely to contribute to winning the war than more ambitious and complicated long-term developments. A comparison of the little 4.5-in. 29-lb barrage rocket and the fearsome V–2 as to their relative effects on the outcome of World War II will show that this conviction has been vindicated.

3. *Safety.* It was always insisted that the designs be thoroughly safe and dependable. This was done not only with a view to preventing casualties among our own men, but also because of a realization that rockets were new to the Services and a poor showing at the beginning might prejudice their users against them and seriously retard their growth into a significant factor in the victory.

Experimenters who come afterward, who have access to many kinds of propellant with diverse properties, who have time to tackle problems of greater difficulty and solve them with greater elegance, and who, having customers eager for their products, may be able to design to smaller safety factors in the interest of obtaining the last ounce of performance, will certainly do things differently. This fact should be kept in mind in reading the following pages.

The author joined the rocket group in June 1942, just as the first American rocket was starting into combat. As a member, and later an assistant supervisor, of the projectile group, he had first-hand experience with most of the rockets discussed in these chapters and hence can reasonably hope that most of what he has written is true. Nevertheless, because of the pressure under which these chapters had to be written and the unavailability of people and information after the development activities ceased, this summary is much more of a one-man job than is desirable for a work of its kind. It is therefore hoped, but not expected, that the number of errors may be few and that the subjects which the author was not directly concerned with during World War II may have been given their proper space and emphasis.

[a] Assistant Supervisor, Section I (Rocket Design and Development) of Contract OEMsr-418 at the California Institute of Technology.

[b] These items head the list of OEMsr-418 reports in the general bibliography in the appendix.

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Chapter 14

MILITARY NEEDS WHICH ROCKETS CAN MEET

By *C. W. Snyder*

14.1 GENERAL CHARACTERISTICS AND USES

During World War II, short-burning, solid fuel rockets were developed to meet many tactical needs. It appears that the field for military application of such rockets has been fairly well explored. Most of the applications for which rockets have been found advantageous have been rather specialized; solid fuel rockets have supplemented shells and bombs rather than displaced them. Before a rocket is chosen or designed for any application, therefore, it should be established that the rocket promises definite advantages over other types of projectiles. For many common tactical situations it does not. For some others, rockets may be the only possible answer or the most effective one.

All actual or proposed uses of rocket projectiles which have come to the attention of the writer involve the familiar functions of shell and bombs, namely, the delivery of materials to the enemy, sometimes at velocities adequate for penetration of his defenses. In addition to solid shot, the materials carried have included high explosives, chemical agents (gas, smoke, incendiary mixtures, etc.), illuminating flares, and certain inert fillers like antiradar "window" and propaganda leaflets.

The principal characteristics of rockets which affect their employment are

1. Their lack of recoil. This is unquestionably their most important advantage and is a factor in nearly all tactical uses.
2. Simplicity, light weight, and associated mobility of rocket launchers as compared to guns.
3. Low setback forces resulting from the usually prolonged period of propulsion.
4. Long, stable underwater and underground trajectories in the case of most fin-stabilized rockets.
5. Superior accuracy and penetrating power of rockets as compared to bombs.

Among the characteristics of rockets which have limited their use are blast, smoke (in some cases), and the effects of temperature on performance. Blast is a hazard and, like smoke and the muzzle flash of guns, reveals firing positions. As a result of developments toward the end of World War II, temperature effects are now much less restrictive.

Largely because of the properties enumerated, the principal tactical uses for which rockets have been preferred over shells and bombs are the following:

1. Firing heavy projectiles from shoulder launchers, small surface craft, light vehicles, and, perhaps most important, airplanes.
2. Drenching area targets with intense barrages for short, though usually critical, periods.
3. Firing from ground locations to which transportation of guns capable of comparable effects is difficult or impossible.
4. Attacking underwater targets like submarines or ship hulls and underground targets like caves.

The tactical situation in view will usually indicate roughly the specifications to be met as to range, velocity, dispersion, weight of payload, total weight, fuzing, and type of launcher. In general, it is desirable to provide launchers to fit the final rocket design, but frequently considerations of launchers already available or of available sizes of tubing from which to make rockets limit the choice of calibers. The following sections show in general terms what combinations of some of these factors can be met with conventional solid fuel rockets. Later chapters cover rocket principles, design, and performance in greater detail.

In addition to their uses as parts of projectiles, solid fuel rocket motors have found employment as thrust units for assisting the take-off of airplanes and of long-range jet-propelled missiles, with and without wings, for propelling oversize fins through the air as targets for antiaircraft gunners, and for projecting lines, cables, and nets for clearance of land mines, and for other similar uses. However, this and the following chapters will be concerned only with rocket projectiles, and mainly with those types developed during the years 1941 to 1945 at the California Institute of Technology under Division 3, Contract OEMsr-418. Most of these rockets were adopted by the Army or Navy, or both. Ballistite, the double-base composition used in trench mortars, was the propellant used in all of these. With the exception of "Tiny Tim," the 12-in., 1,200-lb aircraft rocket, all of them used single-grain charges.

ranges by better streamlining. Another requirement for very long ranges is the extension of the propulsion phase, that is, of the burning time of the propellant. A fuller discussion of the range problem is given in reference 1.

A more practical question than that of the ultimate range is that of the range variation with payload. Figure 1, taken from reference 2, summarizes the data on this point, comparing service rockets with fixed and semifixed shells for howitzers. It is apparent from the figure why rockets have not been used to a significant extent for ground or sea firing at ranges beyond 5,000 yd.

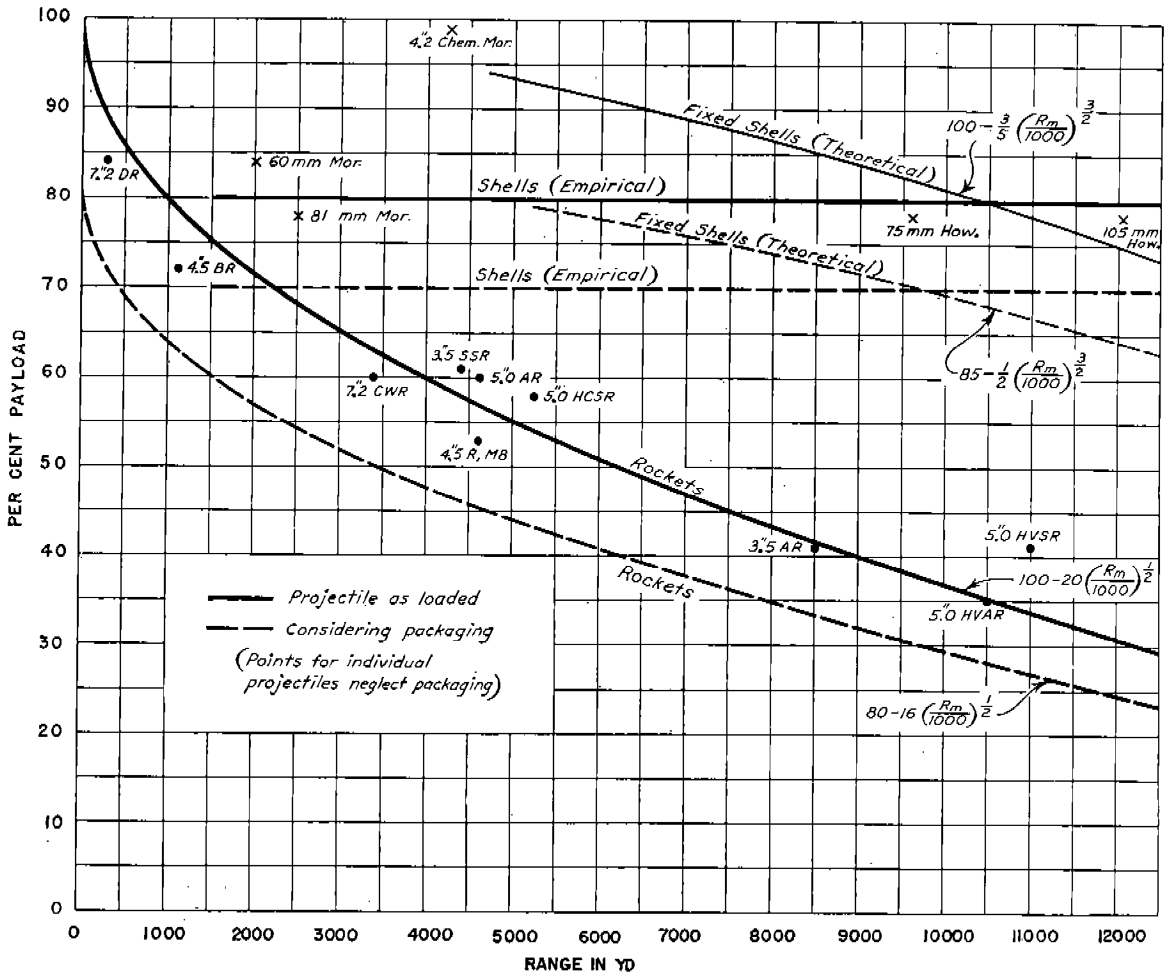

FIGURE 1. Payload vs range.

14.2 RANGE

The maximum range of the rockets considered here is not much greater than 10,000 yd. Attainment of very long range in ground firing is primarily a matter of minimizing supersonic air drag and secondarily one of maximizing the velocity at the end of the propulsion phase, since this phase is a small part of the trajectory length. This point is expanded in Section 21.2. Figure 4 of Chapter 21 shows the effects of air drag and initial velocity on range. CIT put little effort into attempts to extend

14.3 VELOCITY AND PAYLOAD

For a fixed weight of payload (head), the velocity, and hence the range, of a fin-stabilized rocket can vary between wide limits, depending on the size of the motor or, ultimately, on the amount of propellant in the motor. Chapter 22 discusses briefly the limits on the amount of propellant which can be put into a fin-stabilized rocket motor of a given diameter. Theoretically, the problem of attaining maximum velocity is slightly different from that of attaining maximum propellant weight [a] because, as is apparent qualitatively from Figure 12 of Chapter 22, the use of a thicker web than that corresponding to the heaviest possible grain allows

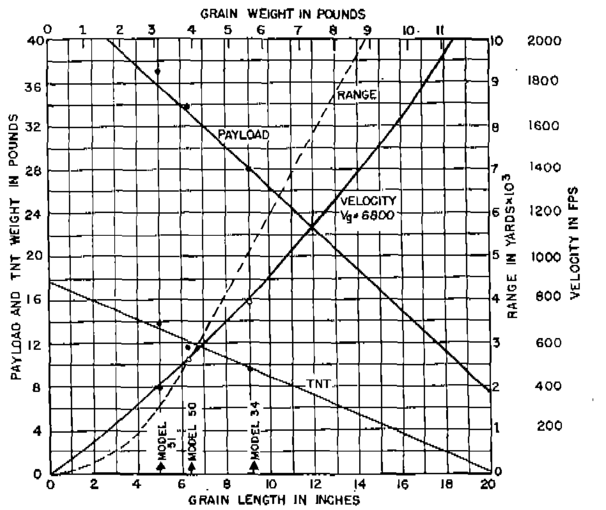

FIGURE 2. Maximum payload and velocity for a series of 5.0-in. high-capacity spinners with cruciform grains.

a considerable reduction in motor length, with a consequent weight reduction which more than compensates for the decreased propellant charge. In practice, however, when factors of propellant strength as well as geometry are considered, the shorter, thicker grains turn out to be preferable even from the standpoint of maximum loading density. Hence, for fin-stabilized rockets, once the maximum grain weight has been determined, the velocity attainable with a motor of a given caliber depends only on the total weight of the rocket, being in fact inversely proportional to it. One can attach to the motor a payload as large as he likes if he accepts the inevitable reductions in velocity and range. The highest velocity so far achieved in a fin-stabilized service rocket is the 1,360 fps of the 5.0-in. *high-velocity aircraft rocket* [HVAR]; this carries a 48-lb head.

With spin-stabilized rockets there is much less freedom in the choice of payloads and velocities, because this type of stabilization imposes rather rigid restrictions on the ratio of length to caliber. With few exceptions, heads and motors of spinners have been of approximately equal diameters. The relationships between velocity and payload are well illustrated in the family of 5.0-in. *high-capacity spinners* [HCSR] developed by CIT. All members of the family have the same diameter, length, and total round weight. Increases in weight and length of the head are associated with corresponding decreases in the motor and in the velocity, as shown in Figure 2. This illustrates the stringent restrictions on possible spinner performance. Thus a high-capacity spinner with the velocity of the fin-stabilized HVAR (1,360 fps) would have a payload of less than 20 lb, and to match the HVAR's 48-lb payload is not possible at any velocity. The spinner could, of course, do a little better with a payload of higher average density. The comparison is shown in a different way in Figure 3 which assumes that 24 lb is the maximum amount of propellant which

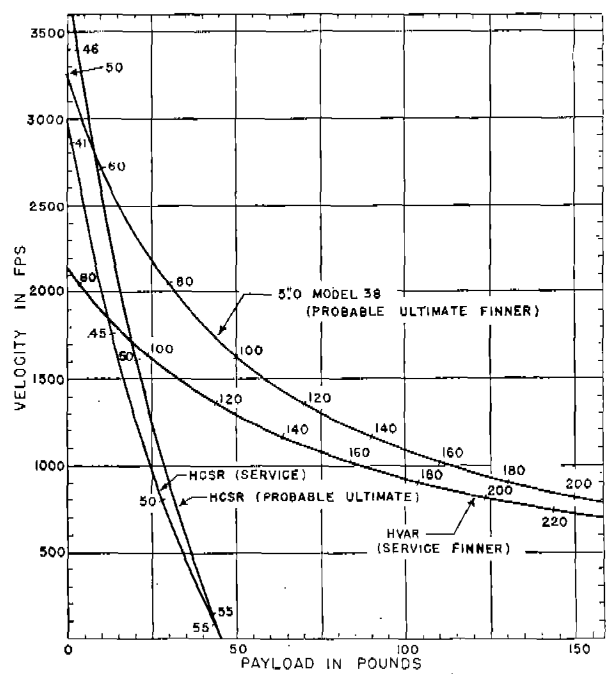

FIGURE 3. Payloads and velocities for 5.0-in. spinners and finners.

[a] A fuller discussion of this point is contained in reference 3, which gives curves for determining graphically the grain configuration which will give maximum velocity for any motor weight and payload.

can be put into a 5.0-in. spinner motor. The curves show that it is only for small payloads that spinners are useful, but their variation of velocity with payload is so steep that in the limit of very small payload they surpass the finners which have more excess weight to carry.

ing, as a function of payload, the velocity which one could reasonably expect to attain with rocket motors of various diameters, but this has not been possible so far because not enough information is available on the variation in motor weight with diameter. The nearest approach that can be made

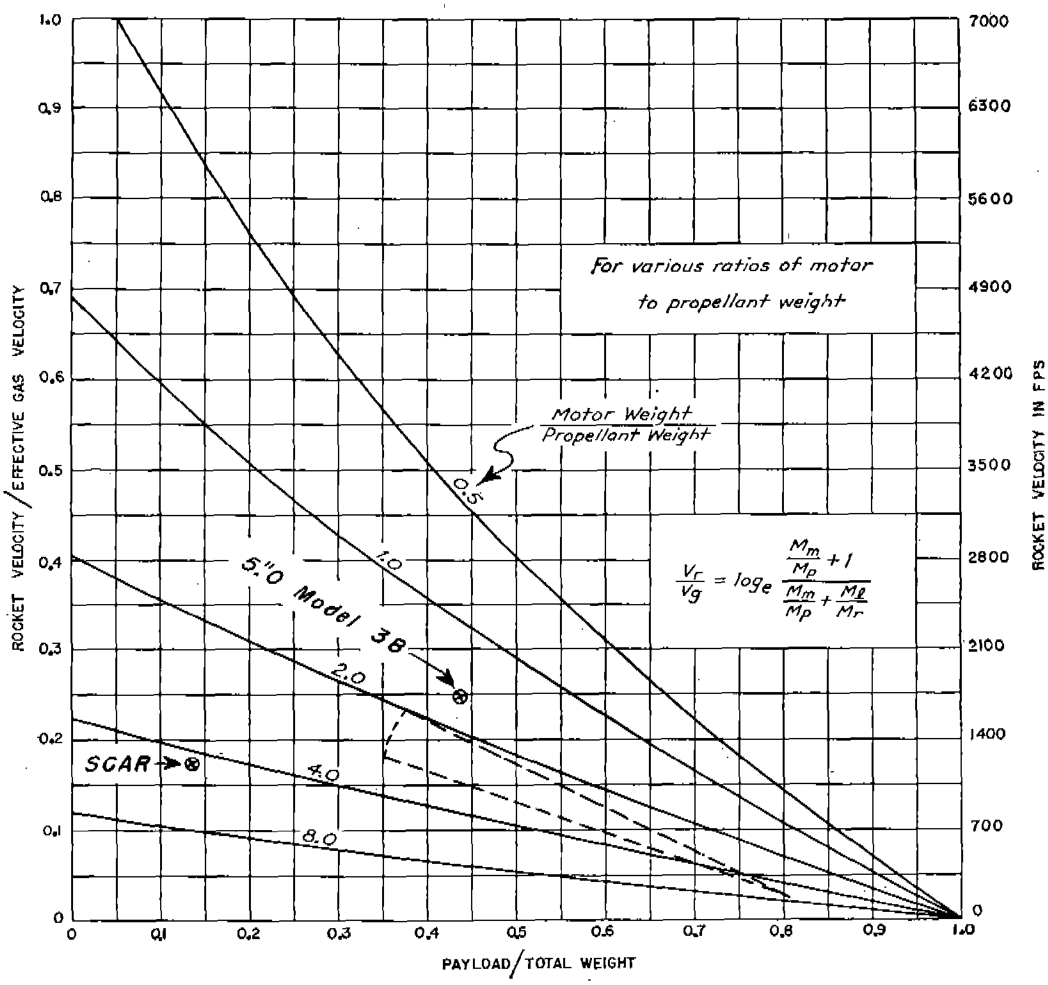

FIGURE 4. Rocket velocity vs payload. With the two exceptions noted, all CIT service rockets fall within the triangular area included inside the dashed line.

If it is desired to impart a certain velocity to a certain payload irrespective of the motor diameter, this can, of course, be done with either a spinner or a finner, but the latter can have a smaller diameter. Because of the marked increase in fabrication difficulty with increasing diameter, particularly for the propellant, one will not choose the spinner unless it has distinct advantages from some other point of view.

It would be useful to compile a set of curves showing, as a function of payload, the velocity which one could reasonably expect to attain with rocket motors of various diameters, but this has not been possible so far because not enough information is available on the variation in motor weight with diameter. The nearest approach that can be made is shown in Figure 4, which illustrates clearly the importance of minimizing motor weight if high velocities are desired. Thus, regardless of how small the payload is, no rocket of any diameter can exceed 2,800 fps unless its ratio of motor weight (including propellant) to propellant weight can be brought below 2.0. This ratio decreases with increasing diameter; for service finners its value averages 5.0 for 2.25-in. motors, 3.0 for 3.25-in. motors, and 2.7 for 5.0-in. motors, but extrapolation

[redacted]

beyond this point is extremely uncertain.[b] One spinner (the 5.0-in. Rocket Mk 7 Mod 0) actually reached 2.0 and has the highest velocity of any CIT service rocket—1,540 fps. With two exceptions, all CIT rockets fall within the narrow triangular area marked off in Figure 4. The exceptions are the 2.25-in. *subcaliber aircraft rocket* [SCAR], which is not strictly comparable with the others because it carries no payload, and the 5.0-in. Motor CIT Model 38 (assumed to have the same payload as the HVAR), which was deliberately designed to have the lightest possible motor by accepting a lower safety factor (narrower temperature limits) than that of the 5.0-in. HVAR and other service rockets. It is important to note that the ratio, motor weight to propellant weight, is directly proportional to the test pressure (i.e., operating pressure times safety factor) and inversely proportional to the tubing tensile strength for any caliber of motor, neglecting heating effects.[2a] (See Chapter 23.) Hence it is not possible to design an efficient rocket falling far outside the triangular area in Figure 4 unless one employs lower safety factors, lower operating pressures, or higher tensile strengths than have been customary, or unless one goes to interior-burning grains so that the use of light metal alloys for motor tubes is possible. (See Section 23.2.6.)

14.4 ACCURACY

The factors determining a rocket's dispersion are relatively involved and are discussed in Chapters 24 and 25. Without attempting to indicate the reasons, we can summarize the dispersions attainable with various types of rockets as follows:

1. *Low-velocity (700 fps or less) fin-stabilized rockets fired from typical stationary launchers.* With burning time (duration of thrust) of approximately 0.5 second, dispersion will be large—well above 20 mils and perhaps above 30. It can be decreased by decreasing the burning time, however, and hence will vary markedly with temperature. If burning times are brought down to 0.1 or 0.2 second as by use of thin-web grains, dispersions less than 10 mils are attainable. If all the burning can be made to take place on the launcher,[c] the dispersion will, of course, be only 2 or 3 mils. The short burning times are feasible only with small payloads or small velocities if single-grain charges are used.[d]

2. *High-velocity (700 to 1,400 fps) fin-stabilized rockets fired from typical stationary launchers.* The smallest dispersion obtained up to the present with conventional designs is just under 20 mils. No means are now apparent for improving this in service rockets. This dispersion is lower than that of comparable slower rockets primarily because of the greater length of the faster rounds. Longer burning times are usually required for the higher velocities, but at these velocities changes in the burning time have little effect on dispersion.

3. *Ground-fired spinners.* Spinners to be fired at high quadrant elevations at ground targets must have relatively low stability in order to follow the curved trajectory; they have a minimum dispersion of slightly under 10 mils and frequently average almost 20 mils at high angles. Five mils or less is attainable with high-spin rockets which are restricted to flat trajectories,[4] but only with extreme care in manufacturing the parts.

4. *Forward-fired aircraft rockets.* Fin-stabilized rockets have ammunition dispersions (exclusive of dispersion due to pilot, plane, wind, and sight) of 2 to 5 mils, with the lower values corresponding to higher aircraft speeds. Spin-stabilized aircraft rockets had not been tested very extensively before the end of World War II, but dispersions of approximately 5 mils laterally and 2.5 mils vertically were being obtained.[e]

An indication of the relative accuracy of rockets and shells is given by Figure 5, taken from reference 2, in which a fuller discussion is contained.

14.5 CHOICE OF FIN OR SPIN STABILIZATION

A first and basic decision which must be made in designing a rocket concerns its type of stabilization.

[b] The ratios quoted are all for motors with single-grain charges. The 11.75-in. motor for the "Tiny Tim" aircraft rocket employed a four-grain charge. With the 18-in. extrusion press being completed at the Naval Ordnance Test Station, Inyokern, California, it will be possible to produce a single-grain charge for a motor of this size. With conservative design, an octoform grain of probably 175 lb could be accommodated. With the present charge support eliminated and with the use of lightweight fins, the loaded motor would weigh only about 330 lb, giving a ratio, comparable to those above, less than 1.9. Still lighter motors may be practicable.

[c] As in the bazooka.

[d] With multiple-grain charges, larger loads can be given higher velocities, but only by accepting higher motor weights.

[e] This development was continued under the Bureau of Ordnance.

The relative advantages of the two types can be summarized as follows:[f]

1. *Simplicity and cheapness.* A given impulse can be obtained with a fin-stabilized motor having a considerably smaller diameter than the necessary spinner motor. Because slim grains are cheaper than fat ones, small tubes more easily machined than large ones, and single nozzles cheaper than canted multiple nozzles, almost any rocket job can be done more cheaply by a finner than by a spinner. Also, launchers for finners are usually lighter and less complicated.

2. *Payload.* For a given diameter and velocity, a finner can carry considerably more payload than a spinner because of the absence of a length restriction. Hence, if the caliber is fixed, there are many rocket jobs which cannot be done by spinners at all.

3. *Accuracy.* Except in the limited region where very short burning times can be used, greater accuracy is attainable with spinners.

4. *Handling.* Their stubbiness and lack of projecting fins makes spinners more easily handled. In particular, they are easily adaptable to automatic launching, as finners are not unless the velocity is so low that a motor of diameter approximately half that of the head or less can be used with a ring tail (e.g., 4.5-in. barrage rocket and 7.2-in. antisubmarine "Mousetrap" rocket).

5. *Aircraft armament.* For firing forward, finners seem to be slightly more accurate, they can carry larger payloads, and they can be fired from simple, external, low-drag launchers. Spinners, but not finners, are readily adaptable to firing from within the wings or fuselage. For firing in other than the forward direction, only spinners offer possibilities.

6. *Underwater stability.* Finners can be made stable for considerable lengths of underwater or underground trajectory, whereas spinners probably cannot. (See, however, Section 25.9.)

7. *Versatility.* Spinner heads and motors must be matched to each other for each application, one type of round for aircraft use, another for accurate, flat-trajectory ground fire, and a third type for high-angle fire, necessarily less accurate. A single finner motor, on the other hand, can be used with many

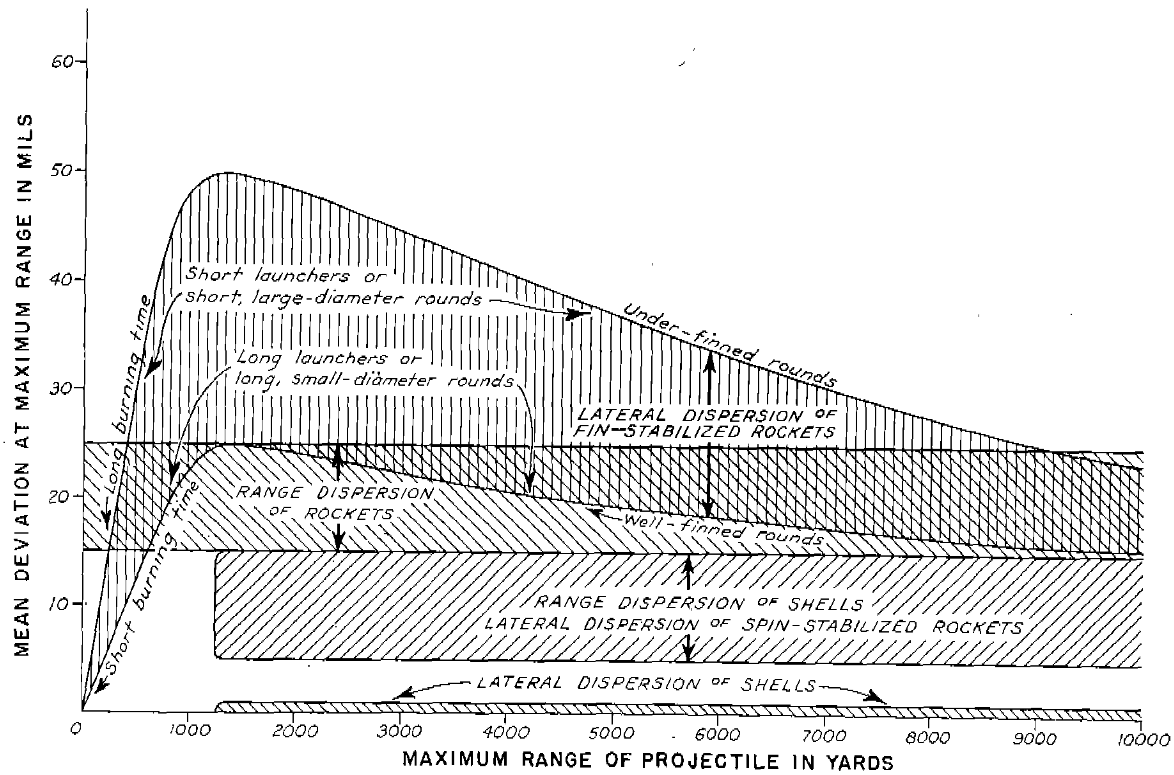

FIGURE 5. Dispersions of rockets and shells.

[f] See also reference 5.

[redacted]

heads for many purposes; in ground fire any of the resulting rounds can be used at all angles of elevation. Most commonly, finners are employed at high angles, with low accuracy, for area barrages.

14.6 EFFICIENCY OF ROCKET ARTILLERY

Questions have been raised frequently as to the efficiency of rockets as compared to other forms of artillery. These questions are applicable, of course, only in those situations in which it is possible to achieve the desired effects at the target with at least one of the other forms of artillery—field guns, machine guns, aircraft bombs, aircraft cannon, etc.—and only when the alternate form of artillery can be made available in the necessary quantity at the necessary time and place.

The efficiency of artillery can be evaluated in various ways. Rockets can be compared (idealistically) with guns in terms of "thermodynamic efficiency," measured by the ratio of the kinetic energy acquired by the projectile to the total energy released by the burning of the propellant. Overlooking heat losses, this reduces to "propulsion efficiency." A simple comparison is that between the amounts of the same propellant needed in a gun and in a rocket to give the same velocity to projectiles of equal masses. On this basis, rockets are considerably less efficient than guns—for example, to give a 25-lb rocket a velocity of 700 fps, 2.5 lb of propellant are required, more than ten times the amount needed in a mortar to fire a shell of about the same weight at this velocity.

An explanation is provided by the principles of mechanics. In each case the energy available from the powder is divided between the projectile and a second agency in such a way that the momentum (product of mass and velocity) of the projectile is equal to and opposite to that of the second agency. With a few simplifying restrictions, we can make the following analysis of gun and rocket action:

$M_1 \equiv$ mass of the projectile.
$M_2 \equiv$ mass of second agency.
$V_1 \equiv$ velocity of projectile.
$V_2 \equiv$ velocity of second agency (in free recoil).
$M_1V_1 \equiv$ momentum of projectile.
$M_2V_2 \equiv$ momentum of second agency.
$E_1 \equiv \frac{1}{2}M_1V_1^2 \equiv$ energy absorbed by projectile.
$E_2 \equiv \frac{1}{2}M_2V_2^2 \equiv$ energy absorbed by second agency.

From the law of conservation of momentum,

$$M_1V_1 = M_2V_2, \tag{1}$$

from which

$$V_2 = \frac{M_1V_1}{M_2}. \tag{2}$$

From the preceding definitions,

$$E_2 = \tfrac{1}{2}M_2V_2^2, \tag{3}$$

so, from equation (2),

$$E_2 = \tfrac{1}{2}M_2\frac{(M_1V_1)^2}{(M_2)^2}, \tag{4}$$

which reduces to

$$E_2 = \tfrac{1}{2}M_1V_1^2\frac{(M_1)}{(M_2)}, \tag{5}$$

and, from the definition of E_1,

$$E_2 = E_1\frac{(M_1)}{(M_2)}. \tag{6}$$

In a gun, the second agency includes all the recoiling components. As the last equation indicates, the energy absorbed by these is less than that given the projectile by the ratio M_1/M_2 of the mass of the projectile to the much larger mass recoiling. Thus most of the energy goes into the projectile. In a rocket, on the other hand, the second agency is the propellant gas ejected at high velocity, and the energy this absorbs is more than that given to the projectile by the ratio M_1/M_2 of the projectile mass to the much smaller mass of propellant. For service rockets the ratio M_1/M_2 varies from 5 to 40, that is, the projectile may receive as little as $1/40$ of the energy available from the propellant.

In the preceding analysis, the gun suffers by the assumption (true for rockets) of free recoil. The effect of restraining the recoil of a gun is to increase the "efficiency" beyond that indicated in the previous paragraph. The amount of propellant required in a gun is proportional to the square of the projectile velocity; in a rocket it is proportional to the first power. Consequently, the apparent "efficiency" advantage of the gun becomes less spectacular at higher velocities. However, as shown in Figure 6, it is maintained well beyond the velocities obtainable with service rockets, with their relatively short burning time. Another factor not covered by "thermodynamic efficiency" becomes more important at the higher velocities; the percentage of payload in the rocket becomes less.

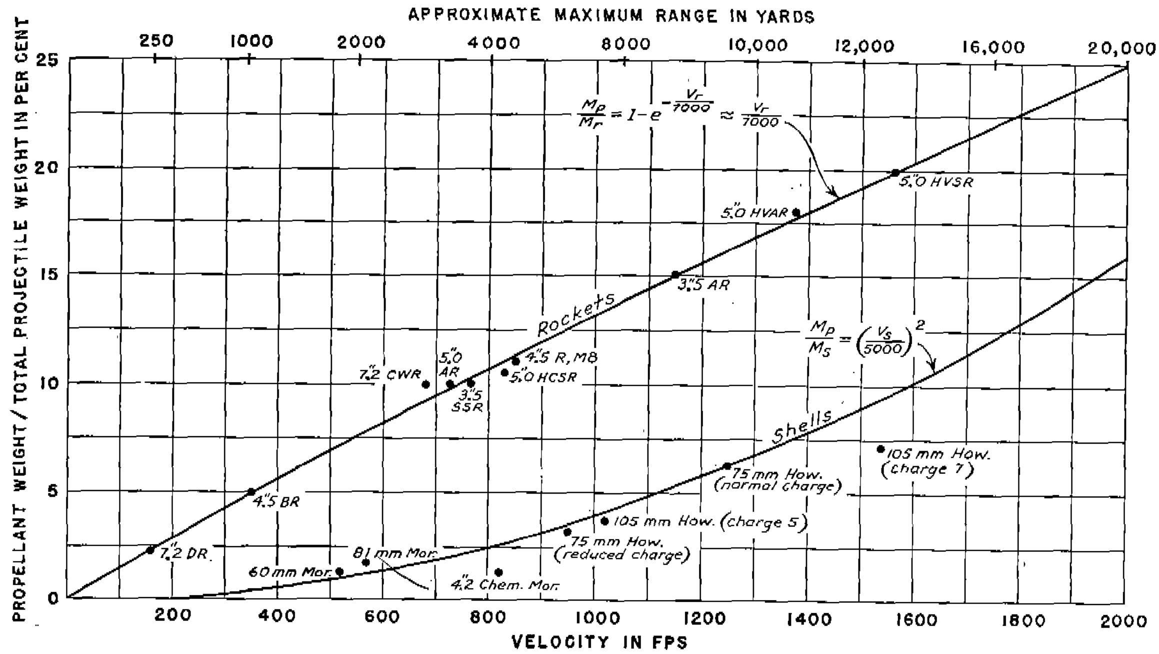

FIGURE 6. Relative propellant weights for rockets and shells.

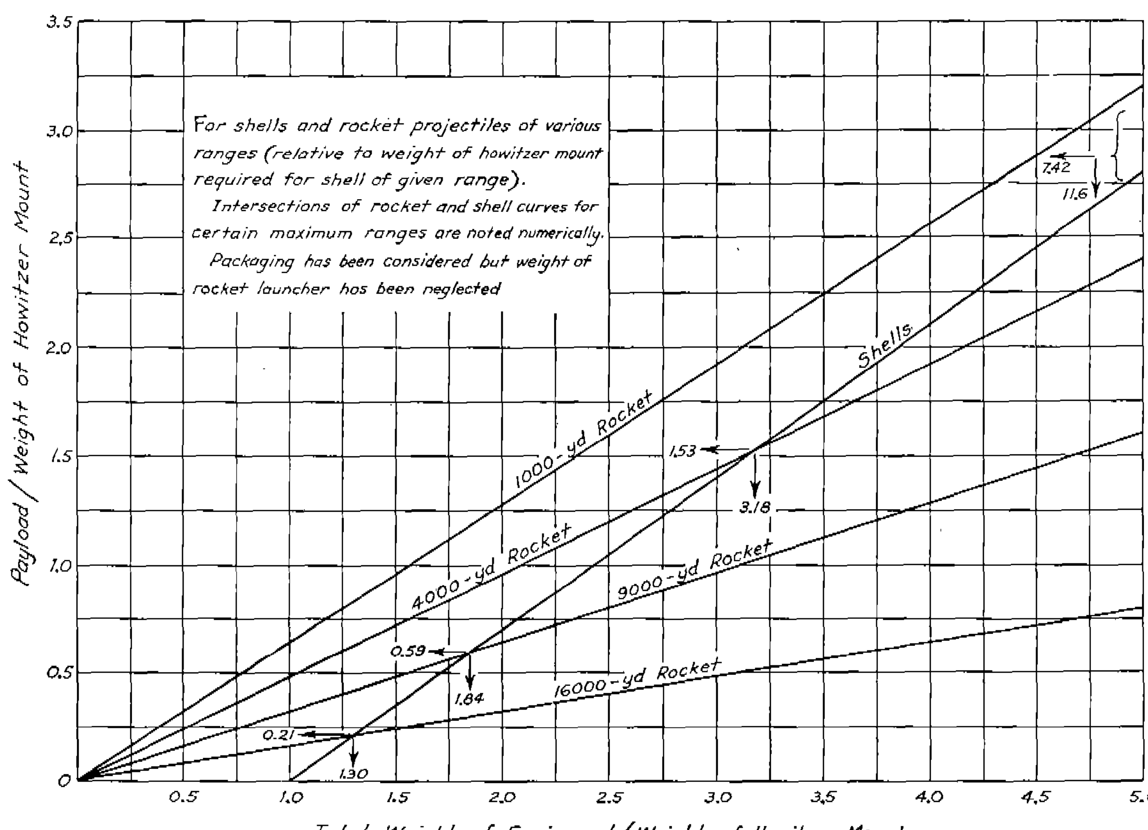

FIGURE 7. Payload vs total weight of equipment.

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More important than efficiency in the use of the energy available from the propellant, from the practical standpoint, is the concept of "military efficiency." This involves a comparison of the amounts of effort required to inflict specified damage on the enemy by various means. This is, of course, an extremely complex problem, but one factor in it can be evaluated by considering the ratio of payload delivered to the target to the total weight of material which is to be transported to the firing point to deliver that payload. In the matter of weight, the rocket has a great advantage because its launcher is so light. The weights of the standard launchers (most of them automatic or multiple) for fin-stabilized ground-fired rockets range from 7 to 37 lb per round. Although an exact comparison with guns involves questions of rate and amount of fire required, the advantage obviously lies with the rocket.

On the other hand, the rocket suffers from the disadvantage that it must carry along its motor, which is usually dead weight from the standpoint of usefulness at the target. This handicap increases with velocity. Hence the velocity or range required affects the choice between rockets and guns as to whether a given amount of payload can be delivered to the enemy with a smaller total weight of equipment. An analysis[2] based on the average weights of various kinds of equipment yields the graphs shown in Figure 7. At the points of intersection (which are marked) between the shell curve and a rocket curve for a particular range, the total amount of equipment necessary to lay a given quantity of effective ammunition (payload) on the target will be the same for both rockets and shells. Below these points, rocket propulsion will be more "efficient." Evidently it is at *short* ranges that rockets have the most distinct advantage, in contrast to the situation for thermodynamic efficiency. During World War II, large numbers of rockets were used for area barrages at ranges from 1,000 to 5,000 yd.

Chapter 15

ROCKET HEADS

By *C. W. Snyder*

15.1 SIMILARITY TO SHELLS AND BOMBS

Rocket heads, exclusive of their fuzes, have been the subject of relatively little experimental investigation. In many cases they have been adapted with relatively minor modifications from standard shells or bombs, which is reasonable since they are intended to do substantially the same job at the target. From the point of view of performance at the target, the problems of exterior contour, optimum well thickness, steel composition and heat treatment, etc., for rocket heads are generally similar to those for the corresponding shells or bombs.

15.2 ALIGNMENT

The relation of the head to the motor does present certain unique problems, of which the foremost is the matter of alignment. The meticulous care which is taken to assure proper alignment of nozzle axis and motor tube to ensure low dispersion is obviously of no avail if the center of mass of the head is far from its axis, so that comparable precautions must be taken in head manufacture. Well thicknesses must be relatively uniform, filling must be symmetrical, and threads for attaching to the motor must be machined so that their axis passes through the center of the mass of the head within the required accuracy. It has been customary to use the same thread specifications on heads as on motors (see under *Alignment* in Section 23.2) although obviously the precision required for head threads depends markedly on the length of the head and its weight relative to the total rocket weight. In any particular case, it is necessary to calculate the effect that various types of head malalignment have on the overall round malalignment and adjust tolerances accordingly. If the head is the major portion of the rocket weight, it may be desirable to balance each one.[a] For fin-stabilized rockets, the goal is to keep the possible overall malalignment of the round under about $^{1}/_{10}$ degree. Although the limit for spinners is not established there are clear indications that dispersions as low as 5 mils (mean deviation) are unattainable unless each main component, and preferably also the assembled round, is dynamically balanced.

15.3 LEAKAGE AND HEATING

The base of the head serves usually as the front closure of the motor chamber; its exposure to the hot gas in the motor creates problems in some cases. Thus, at one time, concern was felt about the heating of the TNT in the head until tests showed that for the short burning times being used $\frac{1}{4}$ in. of steel was sufficient insulation. Inferior steel bar stock may sometimes contain longitudinal "pinholes," however, so that a certain amount of care is still required in manufacturing the base portions of heads and the connectors between rocket motors and heads.

Gas leakage around base fuzes presents a similar problem, which is discussed in greater detail in Chapter 16.

15.4 JOINT STRENGTH

In cases where the motor is required to remain attached to the head after impact, the strength of the joint between the two becomes critical. For this reason the underwater heads for the 3.5-in. aircraft rockets have long "skirts" which extend back of the threads, and the 5.0-in. high-velocity aircraft rocket heads have their connecting threads 3.5 in. forward of the base (see Figure 1). A similar construction

[a] Methods for balancing the heads of fin-stabilized rockets are given in reference 1. Spinner heads require dynamic balancing; equipment for this is discussed in a CIT final report on testing of rockets.[2]

[redacted]

was originally adopted for "Tiny Tim," the 11.75-in. aircraft rocket, but was abandoned for various minor reasons after tests showed that it was not required to prevent breakup on water impact. It may be that it is necessary, however, to minimize the frequency of breakups on ground impacts, so that a redesign will be required if the potentially long underground trajectory of this rocket is to be utilized (see Chapter 24).

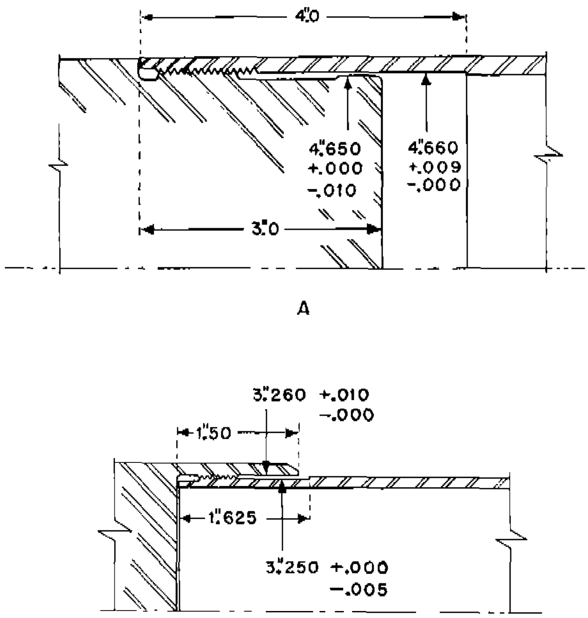

FIGURE 1. Reinforced motor-head connections for aircraft rockets, (A) HVAR, (B) 3.5-in. AR. Diameters of critical surfaces are shown.

16.5 SPECIAL HEAD SHAPES

The shape of a rocket head is usually important mainly from the standpoint of effectiveness at the target, i.e., achieving maximum blast effect or most efficient fragmentation. Occasionally, however, the shape may affect the trajectory to the target. The first case of this kind was encountered with the *antisubmarine rocket* [ASR], which is shown in Figure 1 of Chapter 18. The flat nose of its head was originally copied from the British "Hedgehog" projectile,[b] and extensive underwater trajectory tests at CIT soon demonstrated that it was superior to various other nose shapes suggested because its use resulted in smaller forward travel after impact and less deviation from the mean trajectory. The reason apparently is that its very large drag causes it to be decelerated to less than its terminal velocity during the first 10 ft of underwater travel, after which it sinks almost vertically with increasing speed. The underwater behavior of the ASR and of its cousins the VAR's[c] with various head shapes, tail shapes, fuzes, etc., are discussed in many reports by the CIT Morris Dam group.[d]

The control of the underwater trajectory of aircraft rockets is also a matter of head shape. The fact that fin-stabilized rockets fired forward from aircraft have long, accurate underwater trajectories was discovered by the British, and extensive tests [4] by CIT showed that it was possible by proper attention to head shape to increase the effective underwater range considerably and to introduce a certain amount of control over the curvature of the rocket's path. It is well known that a rapidly moving projectile under water moves in a bubble as illustrated in Figure 2. The water is, of course, held in direct

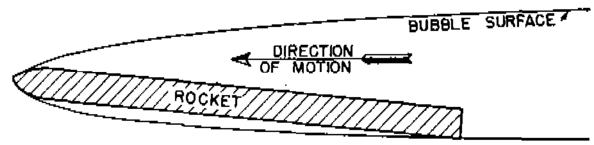

FIGURE 2. Position of rocket under water.

contact with the nose of the projectile, but at some point ahead of the cylindrical portion of the projectile the water recedes from the axis faster than the ogival radius of the projectile increases, so that, in the absence of gravity, the water would touch the projectile nowhere except at the nose. Actually, the rear of the projectile drops to the bottom of the bubble and rides in the water deep enough so that the force of the water on it balances the projectile's weight. Under these circumstances, the resisting force experienced by the projectile depends upon the energy imparted to the water or, in other words, entirely upon the diameter of the bubble and not at all upon the diameter of the projectile. The diameter of the bubble, and hence the resisting force, can be reduced by means of the so-called "double-ogive"

[b] A small spigot-projected antisubmarine depth bomb.

[c] Vertical antisubmarine rockets, also known as retro rockets and retro bombs.

[d] The work of this group is summarized in a CIT final report.[3] In Chapter 1 of this volume Max Mason gives an introductory survey of this work. Reports on it are listed in the CIT OEMsr-418 bibliography in the general bibliography in the appendix.

head; this has a small radius of curvature near the tip of the nose, blending into a curve of much larger radius which joins the straight section at the rear of the head.

Since the rocket travels in its bubble with usually an up yaw, the reaction of the water on the nose is not, in general, symmetrical. An upward force exists which depends greatly on the shape of the ogive at the tip. A hemispherical ogive, since it presents the same appearance to the water even when rotated at a small angle, has almost no upward force. As the ogive is made sharper, the upward force increases. Hence, within the limits of force which the rocket can stand without breaking, one can obtain almost any value of upward force and hence control the curvature of the trajectory by changing the sharpness of the nose.

Three typical heads for the 3.5-in. aircraft rocket, and their performance, are shown in Figure 3. Various other head shapes for 2.25-in., 3.25-in., 5.0-in., and 11.75-in. aircraft rockets are discussed in reports issued by CIT under Contract OEMsr-418.[e]

A rocket penetrates earth or concrete in a manner essentially identical with that in which it penetrates water, so that the theory of head shapes should be the same. This is, in fact, found to be the case, except that the forces in solids are much greater than in liquids so that restrictions on possible head shapes are tighter. Also, as mentioned previously, the strength of the joint between motor and head is much more critical. The underground trajectories of aircraft rockets with various heads are discussed more fully in Chapter 24.

[e] See the CIT OEMsr-418 bibliography in the general bibliography in the appendix.

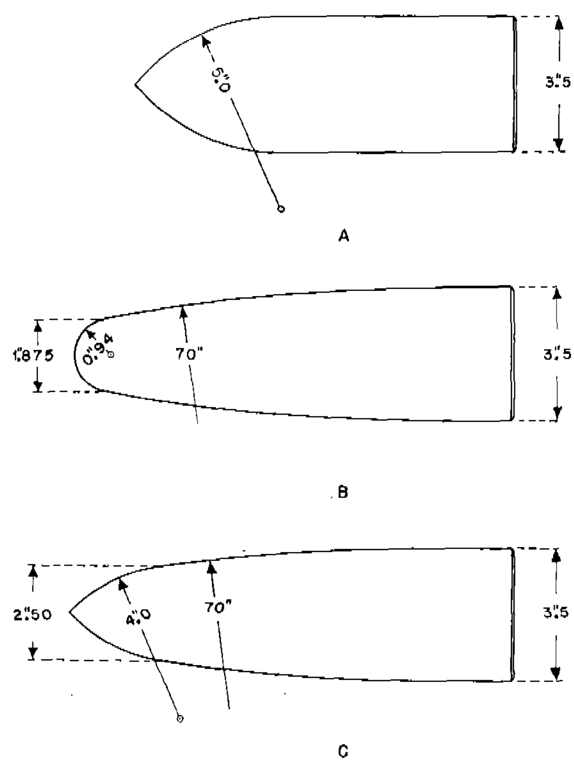

FIGURE 3. 3.5-in. underwater heads.

A. Single-ogive Mk 1
Deceleration coefficient 0.0136
Radius of curvature 200 ft
Distance to half velocity 51 ft

B. Sphere-ogive
Deceleration coefficient 0.0065
Radius of curvature infinite
Distance to half velocity 107 ft

C. Double-ogive Mk 8
Deceleration coefficient 0.0069
Radius of curvature 620 ft
Distance to half velocity 100 ft

[redacted]

Chapter 16

ROCKET FUZES

By *C. W. Snyder*

16.1 GENERAL REQUIREMENTS

Rocket fuzes, like those for bombs and projectiles, have two prime functions: (1) to disperse, ignite, or, usually, detonate the contents of the rocket head under the proper circumstances, and (2) to prevent such actions under all other conceivable circumstances. Because these two basic requirements are distinct, a fuze mechanism can usually be thought of functionally as two sets of interrelated mechanism: (1) the *firing mechanism*, which performs the end functions, and (2) the *arming mechanism*, which prevents firing until completion of a sequence of operations which depend on some of the phenomena associated with the launching and flight of the rocket. Arming is completed when all of the elements in the explosive train, loosely called the "detonator," are uncovered and in line with the firing pin, ready to function on impact or some other stimulus.

Fuze design is a specialized business, consisting mostly of modifications of a relatively few basic types so that they are usable with rockets with drastically different characteristics of pressure, acceleration, burning time, and tactical use. It can only be summarized here, mainly from the more complete discussion of wartime fuze work at CIT given in [a] *Rocket Fuzes*.[1] The following discussion shows how fuze problems may affect the design of other rocket components and indicates the general types of fuzes worked on at CIT.[b] With one exception, all these fuzes are mechanical and differ from standard bomb and projectile fuzes mainly in their methods of arming. With the same exception (the fuze for ejection of "window"), firing of all of these fuzes is accomplished by percussion, by the impinging of a firing pin on a pellet containing a small quantity of sensitive explosive.

The safety requirements for rocket fuzes are substantially the same as those for other projectile fuzes: the arming system should provide restraints on the firing mechanism, and these restraints should remain effective under the forces of transportation, handling, loading, and launching. Many of the rocket fuzes developed during World War II do not entirely satisfy the usual safety requirements.

These requirements are usually more difficult to meet in rocket fuzes than in projectile fuzes, because of the smaller margins between the forces imposed by careless handling and those available for actuation of the arming mechanism. For this reason it is frequently necessary to utilize for arming a combination of forces such that the probability of their simultaneous occurrence under circumstances other than launching and flight of the rocket is negligibly small. In most rocket fuzes, as in projectile fuzes, arming is made to depend on phenomena associated with launching and flight of the projectiles in which they are mounted, and is completed only after a period of projectile flight.

16.2 METHODS OF ARMING

The initiation of the arming process in many fuzes for fin-stabilized rockets, especially those fired from aircraft, depends on withdrawal of a wire, similar to the arming wire used on bomb fuzes. Among the arming methods not dependent on the conditions of rocket launching and initial flight are water pressure, spring-driven flywheels, and deceleration changes. These methods are used only for special target situations. Most rocket fuzes depend for arming actuation on one or more of the following conditions associated with projection; note that two of these conditions are peculiar to rockets.

Acceleration Forces (Setback). In guns, the acceleration of the projectile is very large ($14{,}000g$,[c] for example, in 5.0-in. naval guns) so that setback can

[a] One of the final report volumes issued by CIT under Contract OEMsr-418.

[b] For information on other rocket fuze developments in NDRC, see (1) references 2 and 3; (2) Division 8 Summary Technical Report on fuzing of shaped-charge heads; (3) Division 4 Summary Technical Report on proximity fuzes for rockets.

[c] $g = 32.2$ ft/sec^2 = acceleration of gravity at the surface of the earth. In this example, each element of the projectile is accelerated by a force 14,000 times its weight.

readily be used as a primary arming force. In rockets, on the other hand, acceleration is not only small, being never more than $100g$ in CIT rockets and in extreme cases falling as low as $3g$, but also it varies over a wide range with the propellant temperature. Hence setback cannot safely be used as a primary arming force except in conjunction with other forces. It is often used to delay completion of arming until the end of burning.

Wind Forces. The force provided by the wind streaming past the rocket in flight is most frequently utilized to arm nose fuzes (usually by a propeller) on finned rockets. On rockets having supersonic velocities the air pressure at the nose can be used for arming nose fuzes.

Pressure of the Propellant Gas. Since the pressure in a rocket motor is relatively large, it can be used conveniently to arm base fuzes. The burning times are usually long enough that the entrance of gas into the fuze can be delayed so that arming does not begin until the rocket is well beyond the launcher.

Heat of the Propellant Gas. The hot gas can be used to initiate delay powder trains of special types.

Centrifugal Force. Most shell fuzes make use of the large radial forces set up by the spin, and, since spinning rockets have rates of spin comparable to those of shells, similar or, in some cases, identical fuzes can be used for them. In fin-stabilized rockets, of course, these forces are absent.

The various types of fuzes developed by CIT during World War II were designated by three letters, the first to indicate the method of arming, the second the method of firing (I for impact) and the third the type of projectile (R for rocket in all cases). The following list shows type designations (roughly in the order in which developments started), modes of arming, and the Mark numbers assigned by the Navy Bureau of Ordnance to specific fuzes of these types:

- HIR, armed by *H*ydrostatic pressure, Mks 135 and 140.
- AIR, armed by rotation of a propeller in the *A*irstream, Mks 137, 147, 148, 149.
- SIR, armed by a *S*pring-rotated shaft, Mk 139.
- NIR, armed by air pressure on the *N*ose, Mk 144 (never standardized or used).
- PIR, armed by *P*ressure of the propellant gases in the motor, Mks 146, 157, 159, 163, 164, 165.
- DDR, a special type of the PIR in which firing depends on *D*eceleration *D*iscrimination.

16.3 AIR NOSE FUZES

All nose fuzes which have had extensive use on CIT fin-stabilized rockets have been modifications of the AIR fuze, one of which is shown in Figure 1. At least twelve modifications have been developed

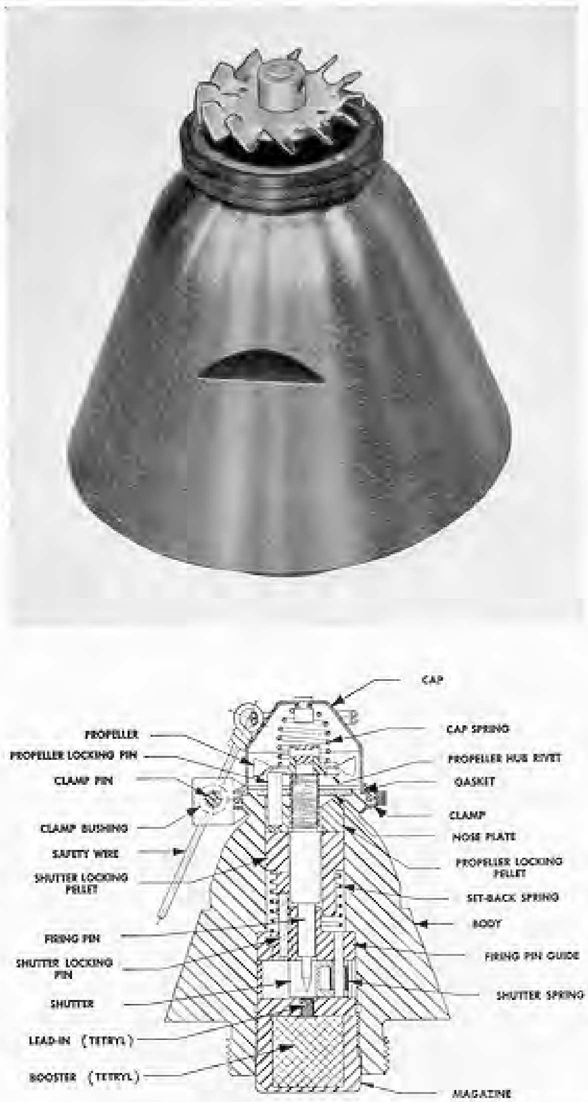

FIGURE 1. Mk 149 (AIR) nose fuze.

for the 4.5-in. barrage rocket, the 7.2-in. chemical warfare rocket, and all the aircraft rockets from 3.5 in. up. In all of them, (1) acceleration retracts a setback block, thus unlocking a propeller which, driven by the airstream, turns a shaft in threads to free the firing pin and complete the arming, and (2)

[redacted]

firing, instantaneous in most cases, is by percussion on impact with the target; the fuzes are point-detonating.

The arming characteristics of AIR fuzes are fitted to the various rockets primarily by varying the size and shape of the propeller and the pitch of its blades. For use on aircraft rockets at high velocity, the Mk 149 fuze (AIR 8) has a streamlined body, and it is protected from corrosion or from the possibility of being fouled with ice by enclosing the propeller in a metal cap which is thrown off by a spring on removal of an arming wire at the time of firing. For use on rocket heads shaped for good underwater trajectory, the AIR 12 has a roughly hemispherical body. The AIR 9, 10, and 11 had special propellers and delay detonators to make them "water-discriminating," to fire on or after underwater hits against ships. Because of the difficulty of keeping a nose fuze intact long enough for it to operate with a delay on armor plate of appreciable thickness, work on these models was finally abandoned in favor of the DDR base fuze, described in Section 16.6.

16.4 NIR NOSE FUZES

Fairly extensive experiments were made with a drastically different type of point-detonating nose fuze, the NIR. Air, compressed in front of the rocket, enters through ports at the nose into a chamber around a sylphon bellows (see Figure 2), which collapses and retracts the firing pin, thus allowing the detonator shutter to move into place and complete the arming. Development work on this fuze was not completed because it was found that, at subsonic velocities, the nose pressure was not enough greater than atmospheric to allow the fuze to function with sufficient reliability. For supersonic velocity, however, the NIR should be reliable, and it is suggested in *Rocket Fuzes* [1] that its use would have distinct advantages in the following cases:

1. For high-velocity aircraft rockets, since the partial arming distance could be increased above that obtainable with AIR fuzes;
2. For slow-spin fin-stabilized rockets with a high velocity, since the spin is great enough to prevent the proper functioning of a propeller-arming fuze but not sufficient to allow arming by centrifugal force;
3. For spinners which attain their full spin velocity on the launcher; and
4. For antisubmarine rockets (arming by hydrodynamic pressure).

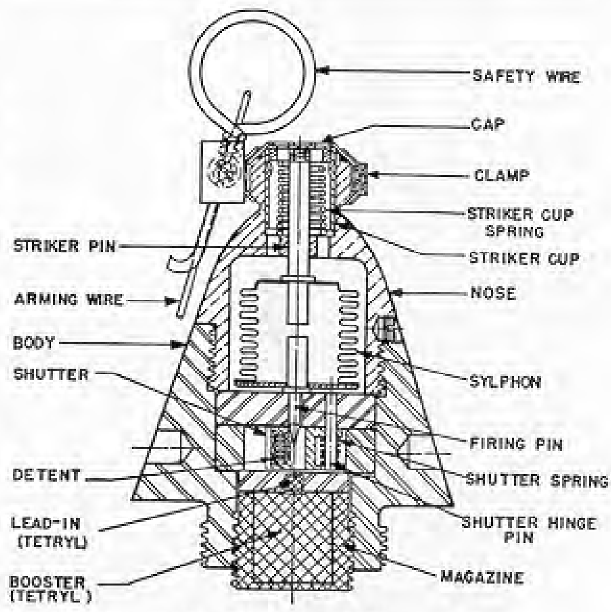

FIGURE 2. Mk 144 (NIR) nose fuze.

16.5 PIR BASE FUZES

Mechanical base fuzes are used in preference to nose fuzes chiefly when it is desired that the head detonate after impact with a certain delay so that

[redacted]

it will penetrate armor instead of blowing up outside it. Base fuzes are fired by the inertia of certain of their parts which tend to continue in motion when the head is decelerated by impact. A small delay (up to about 0.015 second, the exact amount depending on the fuze design and the resistance of the target) is thus inherent in their construction, but, if longer delays are required, they can be achieved by including a pyrotechnic delay in the firing train (0.02-second delay has usually been used) since the head itself protects the fuze from being crushed before the delay element can function.

Most of the mechanical base fuzes which have been used in CIT fin-stabilized rockets are modifications of the PIR, one of which is shown in Figure 3. Arming of these fuzes is accomplished in the following manner. The motor gas, filtered free of solid material by the inlet screen (see Figure 3), enters the pressure chamber through the small orifice in the inlet screw, so that pressure in the chamber builds up slowly during burning. The pressure exerts a force on the diaphragm, and, when the force becomes large enough, it shears the shear wire and depresses the arming plunger, releasing the locking ball, which completes the first stage of arming. The second stage of arming is not completed until the end of burning.

To adapt PIR fuzes for use on different motors, the following modifications can be made. (1) The diameter of the inlet orifice can be varied so that the leakage of gas into the pressure chamber will be rapid enough to reach the necessary pressure before the end of burning at all temperatures expected, but slow enough to provide the delay required for safety. Diameters in the range from 0.0145 to 0.033 in. have been used. (2) The diameter of the shear wire can be varied so that the pressure necessary for arming is just slightly less than the motor pressure at the lower temperature limit. (3) The inlet screen and cup may be replaced with other types of filters and shields as may be required to keep the debris and the closure disks in the motor from clogging the tiny orifice. The first motor on which the PIR fuze was used (the 3.25-in. Mk 7 motor) contained a cellulose acetate igniter case, two fiberboard closure disks, and some cardboard sleeves at the front end, the combination of which created a considerable filtering problem. When this was realized, an effort

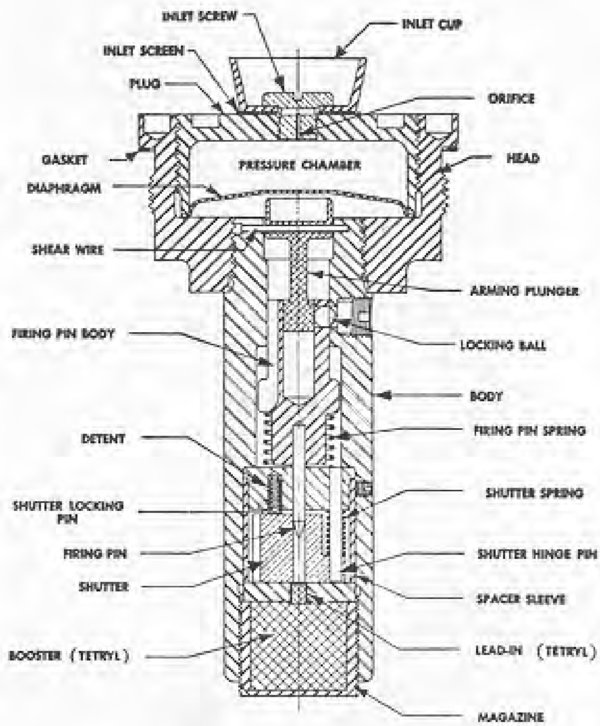

FIGURE 3. Mk 146 (PIR) base fuze.

was made to clean up all motors with which base fuzes were to be used. With the later designs containing metal case igniters and steel closures with "blowout patches" (see Chapter 23), much less clogging has been encountered.

16.5.1 Gas Seals

One of the crucial problems that arises when a base fuze is used is that of making an effective seal between the fuze and the head so that the hot high-pressure gas from the motor cannot reach the high explosive either in the fuze or in the head. The sealing of the inside of the fuze itself is purely a fuze design problem and need not concern us here. For sealing the space between the fuze and the head, early PIR fuzes had a soft copper gasket such as that shown in Figure 3. No particular difficulty with leaks past the gasket had been noted with static firing in connection with fuze testing, but, when a head, in which the base fuze had apparently been omitted so that the gas had direct access to the TNT, detonated low order on the launcher, the whole problem was extensively reinvestigated. The results of this investigation are discussed in the weekly progress reports.[4,5] It was concluded that copper gaskets approximately 0.050 in. thick, annealed soft, provide adequate sealing if (1) the seating surfaces on the fuze flange and in the head are square with the threads, are smooth, clean, and free from defects, and are held within close tolerances; (2) the gasket is in good condition; and (3) the fuze is screwed in with a large torque and seated tightly on the gasket. The tests did not show that prematures could result from leaks such as occur past a poor seal, but it was realized that this is a statistical matter and that even small leaks should not be tolerated. The primary difficulty with the gasket seal is that no way exists by which a bad assembly can be detected after it is made.

As an alternative, gas seals of the type used in gun projectiles were extensively investigated.[5a] In these seals, a copper-encased lead "gas check" is forced into a triangular groove, the sides of the triangle being the edges of the rocket head and the fuze respectively (see Figure 4). Such gas checks were found to be entirely satisfactory if crimped in place with sufficient pressure, even when the parts were poorly assembled or had scratches or gouges on the seating surfaces or threads not at right angles to the seating surfaces. The condition of the gas check as seen on a visual inspection was, within limits, a satisfactory criterion of the effectiveness of the sealing. As a result of these tests, this type of gas check has been adopted for all base fuzes (see Figure 5) except those in which the fuze and the motor adapter are made in one piece so that no space for leakage exists.

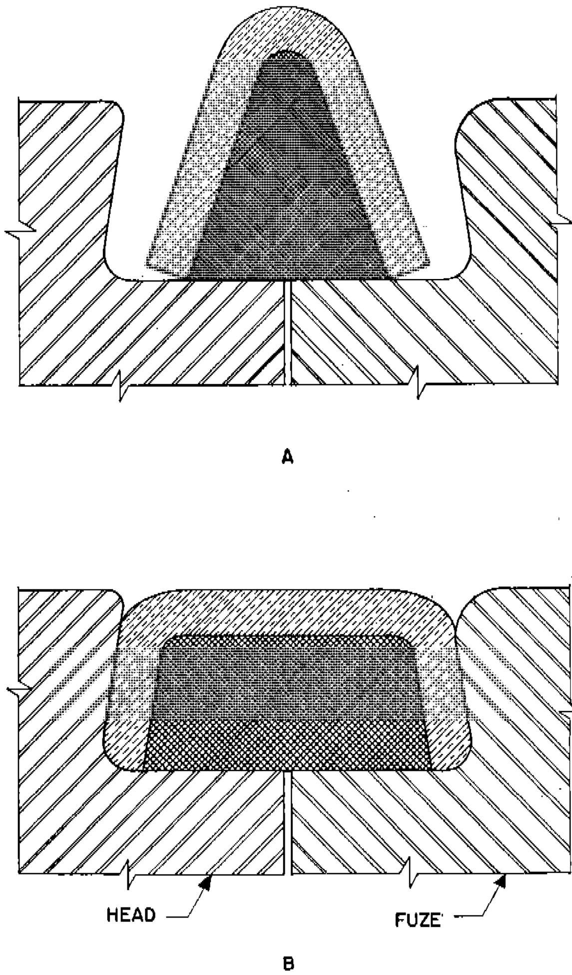

FIGURE 4. Gas check ring (A) undeformed and (B) as actually used. Center is lead and jacket is copper. Illustrations are about 12 times actual size.

16.6 DDR BASE FUZES

As is apparent from Figure 5, the DDR is a modification of the PIR from the standpoint of arming mechanism, but its method of firing is so unorthodox that it has been given a special designation—the "deceleration-discriminating" fuze. It was designed for use with the aircraft rockets which

[redacted]

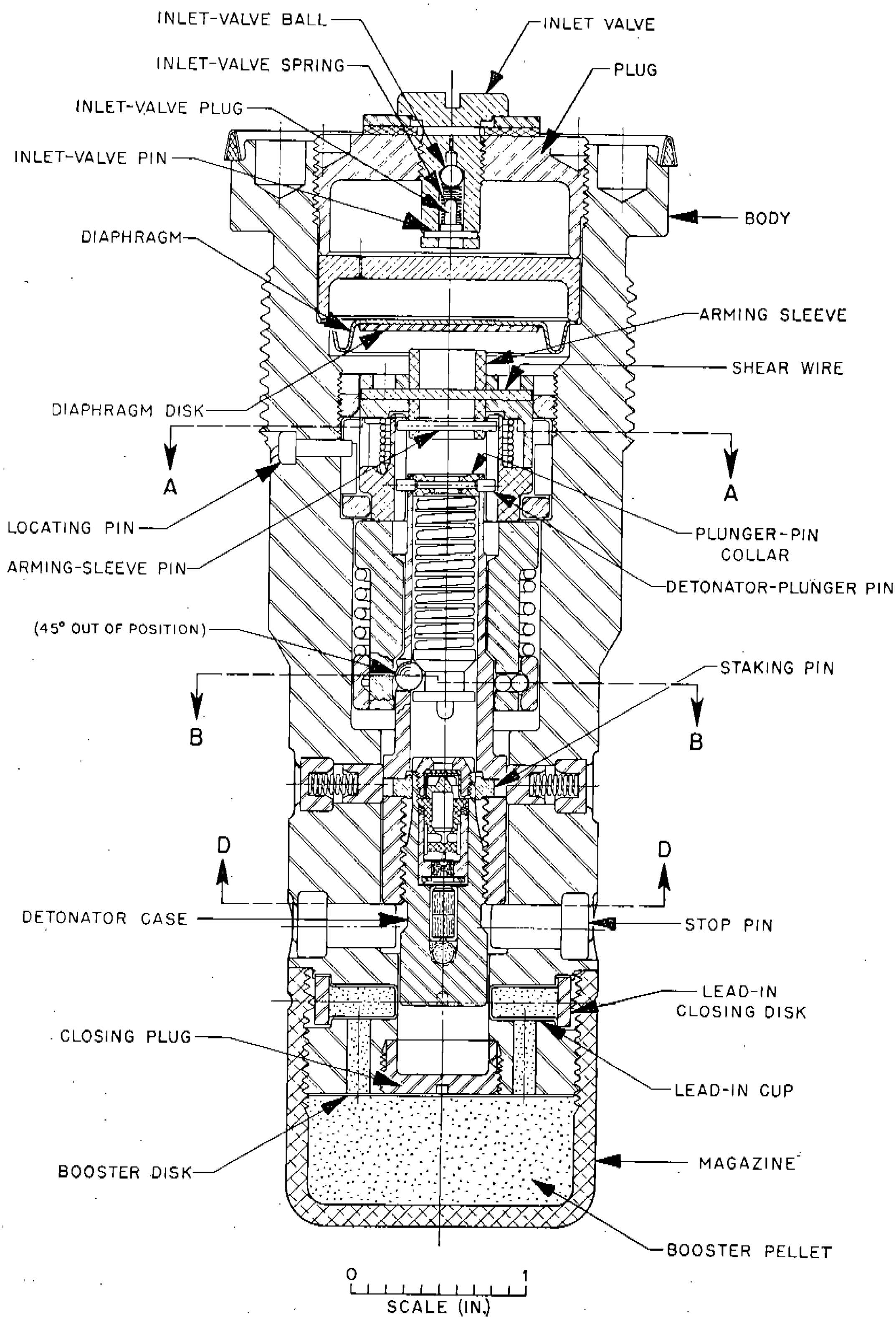

FIGURE 5. Mk 166 Mod 1 (DDR) base fuze.

[redacted]

have stable underwater trajectories, and its operation may be described briefly as follows. The initial impact (on water or target) unlocks a trigger mechanism which is controlled by the deceleration of the rocket. Nothing more happens as long as the deceleration remains more rapid than that which accompanies high-speed underwater travel. Deceleration during armor penetration is, of course, much more rapid than this. When, after exit from the armor, the rocket is traveling through the less resistant air, the slow deceleration causes release of a spring-activated firing pin which initiates the explosive train. Thus the fuze satisfies the basic requirements for the attack of heavy ships—whether the hit is above or below the water line, the fuze detonates after penetration, but does not detonate during impact on the water or on the ship. Since its functioning is independent of time delays and of length of underwater trajectory (within limits), it is effective against armor of any thickness which the rocket will penetrate, and it does not require great precision in the firing of the rocket.

For use against certain land targets such as caves and pillboxes, the DDR fuze has special advantages, since, instead of detonating with a fixed delay after the first impact, it waits until the rocket penetrates the first obstacle completely or is brought to rest in it, thus considerably increasing the destructiveness.[d]

Although the DDR fuze was developed too late to have any service use in World War II, some general remarks about its tactical use can be made.[1a] Obviously a fuze of such unorthodox characteristics will be most effective only under very special conditions. To be useful under water or under ground, it must be used on a rocket which has a stable underwater or underground trajectory and does not break up; the characteristics of such rockets are discussed in Section 24.9. The fuze is rugged and will function after impact at not too great obliquity on fairly heavy plate, so that, if the full potentialities of the fuze are to be realized, the head must be equally rugged. Thus good results were obtained in experimental firings with the 5.0-in. Rocket Heads CIT Model 35 and Mk 2 Mod 2 having solid and heavy noses (adaptations of "special common" type projectiles). The only heads used during World War II, however, for reasons of availability, were modifications of the 5.0-in. Mk 35 AA common shell, which has a hole in the nose and thin walls so that it breaks up on relatively thin plate. In such a head, the DDR would serve no useful purpose.

[d] See reference 6 for discussion of its use in the 11.75-in. aircraft rocket against caves.

16.7 ANTISUBMARINE FUZES

Three fuzes for antisubmarine use on low-velocity rockets were designed by CIT: the HIR or "Hydrostatic-arming, Impact-firing, Rocket" fuze (Mk 135), the HIR 3 (Mk 140), and the SIR or "Spring-arming, Impact-firing, Rocket" fuze (Mk 139). In addition, extensive underwater tests were conducted on the Mk 131 and Mk 136 fuzes, which are two modifications of a British-designed fuze incorporating underwater vane arming and inertia firing, and some redesign work was done on them in the light of the test results.

The two HIR fuzes were very similar in principle, arming being effected by water pressure entering the fuze through ports in the nose and "popping" a phosphor-bronze diaphragm, which, through linkages, unlocked certain restraints and aligned the explosive train. They were fired by the deceleration on impact with a solid object, which released a spring-loaded firing pin. Neither fuze was used extensively in service, since the Mk 131 was simpler to make, was available in quantity earlier, and exhibited superior performance in CIT's underwater tests. Since their operation did not depend on any of the characteristics of the rocket, they are not of particular interest to us here. A full discussion of their design and testing is contained in *Rocket Fuzes*;[1] diagrams and photographs can be found in references 7 and 8. Numerous CIT publications discuss the underwater tests of these fuzes.[9–15]

The Mk 139 fuze, originally designated the SIR, was designed primarily for vertical bombing of submarines from low-flying aircraft. Since the rockets were fired rearward at a speed approximating that of the plane, their flight was somewhat unstable, and the fuzes had to be designed to arm reliably regardless of whether the rocket fell nose down or sideways. It was desired that the fuze fire on contact with the submarine hull either submerged or on the surface, so that water discrimination was necessary. To meet all these requirements, a coiled clock spring was used as the source of arming energy, accelerating a flywheel which gave the arming delay. Water discrimination was achieved by making it point-detonating. The fuze functioned satisfactorily but saw little service use because of the decline of interest in vertical bombing.

No detailed discussion of the fuze will be given here. Information on it may be found in *Rocket Fuzes*,[1] in reference 8, and in various CIT publications.[7,16–19]

16.8 BASE FUZES FOR WINDOW ROCKETS

To eject the payload of antiradar "window" from the heads of 3.5-in. rockets at the proper range and height, a time fuze was required, and the simplest such device appeared to be a powder train in the base of the head, initiated by the motor gases. A percussion-actuated dynamite-fuze ejector unit designated the DU–5 was developed by CIT. Shown in Figure 6, it consists of a plastic case containing approximately 20 g of FFFG black powder, within which is coiled 4½ in. of dynamite fuze (Bickford cord) sheathed in a vinyl chloride tube. The end of the fuze projects through a hole in the end of the case and is cemented in place, and a short length of Quickmatch, held in contact with the end of the fuze by a metal clip, assures easy ignition. Ignition is accomplished by the firing of a .32-caliber blank cartridge containing approximately one-fourth of its normal powder charge. A firing pin attached to a diaphragm sets off the cartridge when the pressure builds up in the motor.

The tests involved in developing the DU–5 are discussed in detail in reference 20. The principal difficulties encountered were:

1. A design of firing pin was required which would not allow gas from the motor to leak into the head through the hole that was frequently opened in the cartridge when it fired. Gas leakage into the head reduced the ejection time because the dynamite fuze burned more rapidly at higher pressure, and in extreme cases the gas pressure blew the payload out of the head.

2. Premature ejections were frequently obtained because the fuze burned through the side and ignited the black powder. This was eliminated by the vinyl chloride sheath.

3. Erratic burning rates of the fuze caused by the building up of pressure from its own gas were eliminated by venting into the payload, which was relatively porous and provided a sufficient volume so that the pressure rise was small.

Although the DU–5 does work satisfactorily and was used in service, considerably better designs

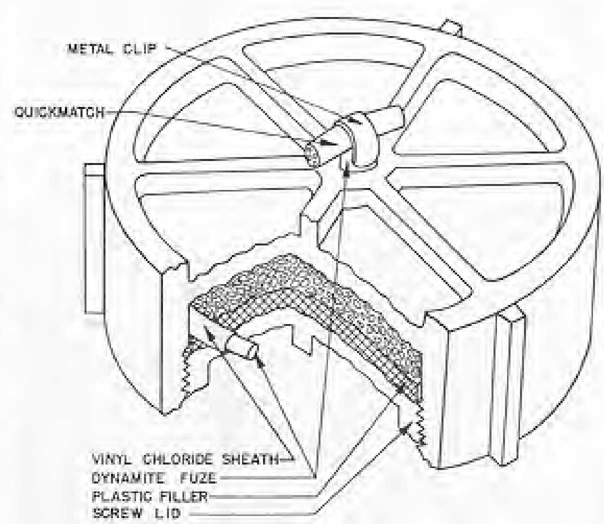

FIGURE 6. DU–5 delay ejector charge for window rocket heads.

from the standpoint of ruggedness, compactness, and simplicity are possible using fuze materials having solid rather than gaseous products. CIT tests on two such units designed by the Catalyst Research Corporation are also discussed in reference 20. Although neither was satisfactory as it was, further research might remedy the defects. It may be possible to initiate the fuze by the heat of the motor gas itself, thus considerably simplifying the design.[e]

16.9 FUZES FOR SPIN-STABILIZED ROCKETS

All fuzes mentioned previously were developed for fin-stabilized rockets. Development work on spinner fuzes was not nearly so extensive because the advent of spinners came fairly late in the CIT work and because standard projectile fuzes, nearly all of which are armed by centrifugal force, can be used with little modification. For the 5.0-in. Rocket Mk 7 Mod 1, the base fuze Mk 31 Mod 0 was used without any modification. Only very minor modifications were made to the Auxiliary Detonating Fuze Mk 44 Mods 1 and 2, which are used with the nose fuzes of all but one of the spinner models developed by CIT. Two point-detonating nose fuzes, the Mk 30 Mod 3 and the Mk 100 Mod 0, have been used on service spinners; both of them are modifications of the Army M48 fuze.

In adapting fuzes to various spin-stabilized rockets, the important factor is to have the arming occur as close to the end of burning as feasible. It may be necessary merely to use a spring with a different tension so that the arming mechanism will be actuated at a different spin velocity. For rockets fired at long range, the spin may drop to 75 per cent of its maximum value, so that, if the arming is reversible (as is the case with both the Mk 30 and Mk 100 nose fuzes), it must take place at less than 75 per cent of maximum spin (corresponding to approximately half the burning distance) if the fuze is not to become unarmed again before impact. Since centrifugal force increases with the distance from the axis, a detent which has moved out and armed the fuze at a particular spin velocity is exerting considerably more force than before. It is therefore possible to arrange that the arming process will not reverse until the spin has dropped considerably below that at which it occurred. The factors involved in obtaining this "unbalanced" condition are discussed in *Rocket Fuzes*.[1b]

[e] Earlier work by the Catalyst Research Corporation with Section H of Division 3 on the development of gasless delay units for ejecting parachute flares from the heads of 3.25-in. rockets is covered in its final report listed in the general bibliography.

In 1942 some development work was done by Section H, working with the Navy at the Naval Powder Factory, Indian Head, Maryland, on a delay-ejection device in which the action was initiated when the propellant gases in a rocket motor heated a metal tube to melt solder within it, to release a pin.[21,22]

Chapter 17

ROCKET LAUNCHERS

By *C. W. Snyder*

17.1 INTRODUCTION

In the development of an effective rocket weapon, the proper design of launcher is no less important than that of the projectile itself. Nevertheless, the space devoted to launchers here will be small because their problems are for the most part almost indistinct from those of the rocket itself and because they are discussed fully in two of the CIT final report volumes.[1,2]

Many types of launchers have been used, varying in complexity from simple cardboard tubes or wooden troughs to elaborate mechanisms for loading, aiming, and firing by remote control. Naturally many considerations enter into launcher design. The starting point is the tactical employment, the round to be used, and the platform or vehicle on which the launcher is to be mounted. These will determine the nature, length, and number of the guides, the nature of the mount, the electrical system, and the type of fire control. Consideration must be given to the control of the rocket blast and to such factors as the means of loading, protection from weather, and limitations on shipping volume and handling weight. These considerations for rockets fired from aircraft differ so radically from those for rockets fired from stationary platforms or surface vehicles that it proved efficient to have two distinct groups to handle the two types of launcher problems. This division will be observed in the following discussion.

17.2 SURFACE LAUNCHERS

In *Rocket Launchers for Surface Use* [1] a thorough discussion of the problems of launcher design for surface-fired finners and spinners is given, with complete descriptions and illustrations of all launchers which saw any service use. We shall not attempt to duplicate the material here.

17.2.1 Launcher Types

The basic function of a launcher is to support and guide the rocket in its initial motion. Three commonly used means are rail launchers, slot launchers, and tube launchers. In the first the rocket slides on two guide rails so spaced as to subtend an angle at the rocket axis of 90 to 120 degrees. The rails are commonly made of formed sheet steel or small-diameter steel pipe. If the launcher is to be used on a moving vehicle, one or two upper rails may be added to hold the rocket down. Because of their small weight, rail launchers have been used widely for the low-velocity fin-stabilized rounds consisting of a head and ring tail (fins) of one diameter and a motor of smaller diameter.

The guide rails are made as long as practicable, to increase accuracy, but seldom more than three times the length of the round. In some cases, launcher length has been combined with ease of handling by the use of folding or telescoping rails.

Many aircraft rockets have lug "buttons" by which they are mounted on slotted launchers. The slot is a space of about $\frac{3}{8}$ in. between two flat rails. A few slotted launchers have been developed for firing these aircraft rockets from ships. An example is the CIT Type 31C (see Figure 6 of Chapter 19) discussed in Section 19.2.5.

Rather long tube launchers have been used for certain finned rockets in which the fin diameters could be limited to those of the heads and for rockets equipped with folding fins. In these launchers the tubes were of the same nominal inside diameters as the rounds. Tube launchers have found even wider use for spinners, for with these the launcher length can be reduced almost to that of the round with little loss of accuracy. The short length makes weight less important. Most of the CIT spinner launchers were tubular, with clearance between tube and round provided by three or four internal guide rails. This type of launcher has given the best accuracy under service conditions.

Single-guide launchers, into which only one round at a time can be loaded, are used for applications where portability is more important than rate of fire. For greater fire power, multiple launchers have been used extensively, with number of tubes or rails varying from 2 to 144. The launcher weight per round is little different from that of the single-shot launcher. A considerable saving in weight and an enormous advantage in simplicity and flexibility is afforded by automatic launchers, which fire many rockets from each guide. For light rockets like the 4.5-in. barrage rockets, simple gravity-fed automatics have displaced multiple launchers in many applications. Their disadvantages are (1) the possibility of interruption of the salvo by one defective round or by improper feeding, (2) a considerable increase in dispersion caused by the effect of the blast of one rocket on the flight of the following one, and (3) a limitation on the quadrant angles at which the launchers will operate. Far outweighing these, however, are the advantages of decreased weight and of standardization; a few miscellaneous fittings enable the same launcher to be used either singly or in multiple from virtually any type of vehicle or ship. The primary application of multiple-guide launchers is for larger rockets or for tactical situations where variable train and elevation are required.[a]

Finally, launchers may be classified by their type of mount, which is determined obviously by the tactical use. In certain cases (for example, the CIT Type 60 32-barrel closed-breech launcher designed for use with 5.0-in. spinners against suicide planes) continuous variation in train and elevation may be required, and some standard artillery mount has usually been used. Such flexibility is usually not essential, however, and in the interest of simplicity it has been the practice to give launchers as few degrees of freedom as possible. Some launchers have fixed mounts, set, for example, to fire at 45-degree elevation and aimable only by turning the vehicle on which they are mounted. Most mounts are semifixed, that is, elevation and/or train may be adjusted before firing, but not during the firing. The required accuracy of adjustment depends on the accuracy of the round and the stability of the firing platform.

17.2.2 Blast

In the design, installation, and use of rocket launchers, blast is usually an important problem. Although the direct blast is confined to a cone narrower than the nozzle exit, it may cover a sizable area at some distance back from the round. Also, the air surrounding the direct blast cone acquires high velocity. On any obstruction large enough to intercept all of it, the blast may exert a force roughly equal to the thrust on the rocket—for example, 20,000 lb for the 11.75-in. aircraft rocket. The blast can also ignite, burn, or scorch objects exposed to it. Hence personnel and equipment, including the launcher itself, must be protected from blast. The simplest way is to locate the launcher where blast need not be deflected, as, for example, at the extreme rear or outboard of vehicles and boats. When this is impossible, simple blast deflectors are used, with small recoil effects. A few closed-breech tube launchers have been used, in which the gas reverses its direction and escapes forward around the rocket. In this case the recoil forces, though substantial, will not usually rival those of an equivalent gun because only a small fraction of the propellant burns while the rocket is in the launcher.

In all cases, launcher parts are (or should be) designed to expose minimum area to the blast, all auxiliary equipment is securely mounted, as far off the rocket axis as possible, and electrical assemblies are completely enclosed.

17.2.3 Firing Systems

Most rockets are fired electrically and require a current of at least ½ ampere for reliable ignition. The components of a firing system are a source of power, a control and distribution panel, and the sockets or contacts on the guides themselves, together with the necessary wiring. Although the design problems are mostly straightforward,[3] they require careful attention, for failure of the electrical system is one of the most common difficulties experienced in rocket installations. A storage battery, magneto, or blasting machine suffices as a source of power.

Since rockets are almost always fired in salvo, a control panel is required. This usually incorporates a safety plug, master power switch, indicator lamp, push-button firing switch, and individual push buttons or a selector switch for the circuits to the launchers. Proper design here is essential to prevent accidental firing. The safety plug is removable and should be carried by the loader while at the launcher. Both it and the firing switch should be double-pole,

[a] The Navy Mk 102 launcher is an example of a powered automatic, with elevation and train continuously variable during firing.

[redacted]

both live and ground leads running through them, so that no short circuit or error in wiring can set off a round.

Wiring and contact problems become more complicated on shipboard because of the possibility of deposition of salt from the spray, which may short-circuit the contacts if they are designed with improper clearance, and because of the existence of stray potentials, sometimes amounting to several volts, arising from galvanic action or from leakage. In multiple circuits with parallel wiring, undesired ignition may occur through "sneak circuits"; the location and elimination of these may be extremely time-consuming. The nature of the firing circuit depends on whether single shots, ripple fire, simultaneous salvos, or combinations of these are required, and the various possibilities are discussed in some detail in *Rocket Launchers for Surface Use*.[1]

17.3 AIRBORNE ROCKET LAUNCHERS

In contrast to the situation for surface launchers, where there is a great multiplicity of designs of a relatively few basic elements, only a few aircraft launchers have gotten beyond the test stage, and there are almost as many basic types as launchers. Consequently most of the design problems have been specific to a particular type. The launchers which have reached service use are described in *Firing of Rockets from Aircraft*.[2] These and one or two others are discussed briefly in the remainder of this chapter.

FIGURE 1. Vertical bombing installation under port wing of PBY-5, loaded with ASR's.

17.3.1 Launchers for Retro Firing

For the attack of submarines from airplanes directly above them (as required by the characteristics of the *magnetic airborne detector* [MAD]) CIT developed a series of *vertical antisubmarine rockets* [VAR] known also as retro rockets or retro bombs. In use, these were mounted under the airplane wings (usually) and projected backward with speeds just sufficient to cancel the forward speed of the airplane, to fall vertically. The launchers adopted for service use with these rockets consisted of channels about 7 in. wide, 2 in. deep, and 8 ft long, fabricated of ⅛-in. Dural sheet. The rockets, with heads and tails 7.2 in. in diameter, were provided with lugs which engaged the lower edges of the channels. An electrically insulated spring latch at the forward end of the launcher made electrical contact with an insulated ring on the tail and held the rocket in place until fired. The launching channels, in the form of inverted troughs above the rockets, provided blast protection for the airplane.

top. Both 4.5-in. and 7.2-in. rockets were tested. The other was a Dural framework for launching 100-lb bombs backward from under the belly of the A–20C, using six 2.25-in. rocket motors for propulsion. Results of tests of these installations are described in the PMC [7] and NMC [8] series of CIT weekly progress reports for the period from October 1942 to May 1943.

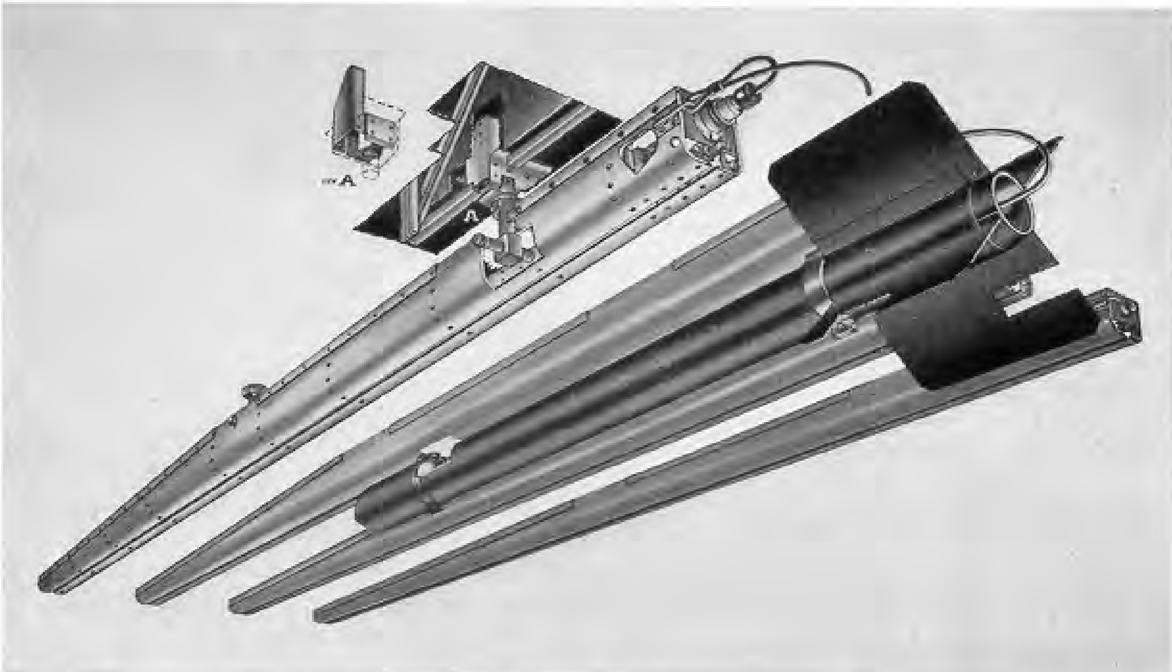

FIGURE 2. Drawing of Mk 4 launcher, showing the method of mounting and the harmonization adjustment. Rocket shown is the 3.5-in. AR Model 1.

The method of attachment of the launchers was necessarily different for each type of airplane. In all cases they were mounted in groups of from 4 to 12 and so oriented as to obtain the desired impact pattern. The only two installations which saw service use were those on the PBY–5 (see Figure 1), which had twelve launchers under each wing, and the TBF–1, which had a total of 8 launchers mounted on the outside of the bomb bay doors.[b]

A considerable amount of experimental work was done on two other installations, neither of which reached the stage of service use. One was an installation consisting of one or more tubes passing through the fuselage at an angle so that the faster rockets were directed downward and backward and the blast came out the open end of the tube at the

[b] Reports on the retro installations include references 4, 5, and 6.

17.3.2 Rail Launchers for Forward Firing

When designs for forward-firing launchers were first discussed, it was realized that finners launched in a high-velocity head wind would have so little dispersion that it might be possible to have the launchers extremely short. The British were using long slotted rails mounted under the wings for forward-firing their 3.0-in. aircraft rockets, however, and had reported trial and abandonment of very short launchers. It was decided that the relatively long British-type launchers held the most promise for first quick development. The result of this decision was the Mk 4 so-called "T-slot" launcher shown in Figure 2. It is a Dural box 90 in. long with a ⅜-in. wide slot running the full length of the lower side to engage the two lugs on the top of the 3.5-in. aircraft rocket. A spring catch at the back permits breech loading and prevents the rocket from sliding out to the rear. Forward of this catch is a shear-wire latch which is held by a copper shear wire strong enough to retain the round during arrested landings but weak enough to shear under the thrust of the rocket when fired. Electrical contact to the round is provided by a two-prong plug at the rear of the rail. Later the length of the rail was reduced to 70 in. for some airplanes.

17.3.3 Post Launchers for Forward Firing

The decision to try rail launchers first was an unfortunate one, because within a year from their first use they were abandoned, leaving the rockets with lug bands which, designed for the T-slot rail, were far from ideal for the launchers subsequently designed. After five months of work on rail launchers experiments with "zero-length" launchers began, and it became evident quickly that the loss in accuracy was only about 2 mils in most cases, not enough to justify the greater weight, complexity, and drag of the rail launchers.

FIGURE 3. Front post of post launcher with Mk 1 HVAR motor attached.

The basic design of all zero-length launchers (later officially designated "post launchers") is the same except for the latch mechanisms. Each launcher consists of two posts under the wing, one behind the other. The short slotted rail of the front one (Figure 3) engages the standard sway-brace type lug band which was originally designed for the long rail launcher; a flat tongue on the rear one (Figure 4) fits into a tunnel-type lug band which is lower than the front lug to ensure clearance of the front post when the rocket is fired.

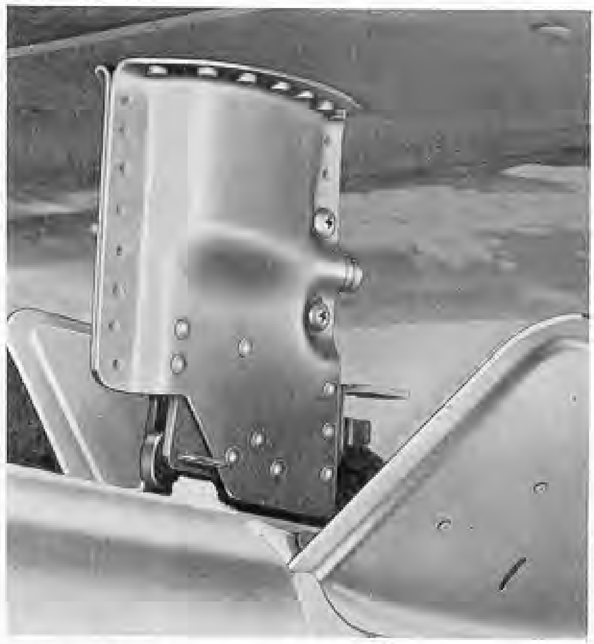

FIGURE 4. Rear post of post launcher with Mk 1 HVAR motor attached and shear wire in place.

The fuze-arming solenoid is usually in the front post and a latch and electric receptacle in the rear. Various types of latches for holding the round on the launcher were designed, and at the end of World War II no decision on the most satisfactory type had been reached. The rocket becomes free immediately after firing, the front lug being guided for only 1½ in. and the rear for about ½ in.

The first standard production post launcher was the Mk 5, die-formed and spot-welded from aluminum sheet. This worked well with the 3.5- and 5.0-in. aircraft rockets (50 and 80 lb) but it was insufficiently strong for the 140-lb, 5.0-in. *high-velocity aircraft rocket* [HVAR]. SAE 4130 alloy steel proved to be the best material and was specified for the launchers designed later for Army planes.

On most airplanes several of these launchers were mounted under each wing in front of the landing flaps, since it was thought that the usual minor blast damage occurring at the trailing edges of the wings could be withstood better by the flaps than by the ailerons, which are weaker and more critical. On the P–38, however, the propeller circles cover almost the entire flap region, and the only space available for mounting launchers was in front of the ailerons, where the problem was complicated by the fact that the wing chord is too small for normal fore and aft separation of vertical post launchers. To meet this requirement, the tree-type launcher (Figure 5) was developed. This consists of one pair of large posts under each wing; each post has branches which permit the loading of five rockets in a cluster.

All these launchers will accommodate the 3.5-in. AR's, the 5.0-in. AR's and the 5.0-in. HVAR's, since these rockets have the same type and spacing of suspension lug. For the 2.25-in. subcaliber practice ammunition, which have lug buttons only about 18 in. apart, a special adapter (Figure 6) was developed which is mounted on the post launcher in the same way as a standard full-caliber rocket, and is held in place by a shear wire which is approx-

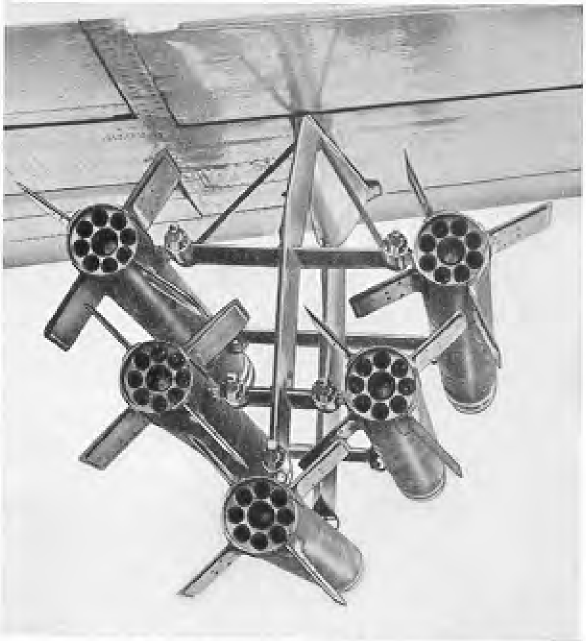

FIGURE 5A. 5.0-in. HVAR's loaded on tree launcher under wing of P-38L.

FIGURE 5B. 5.0-in. HVAR's loaded on tree launcher under wing of P-38L.

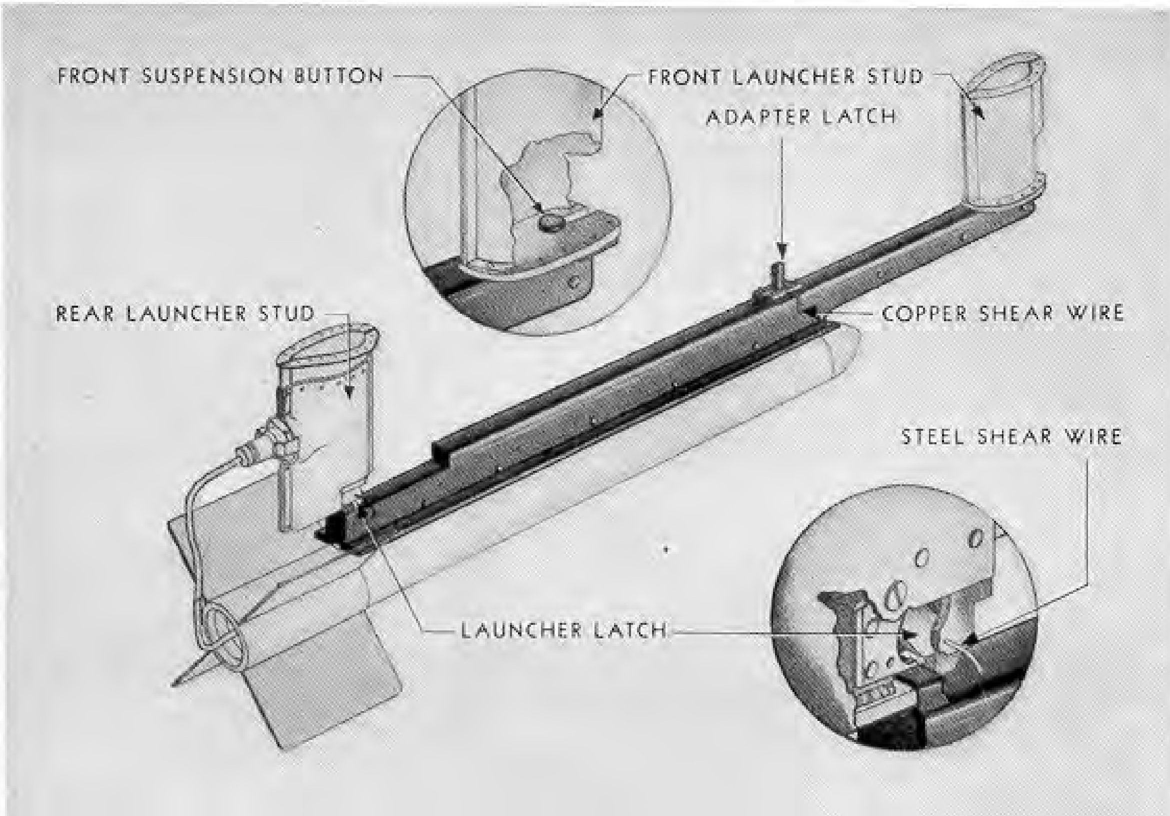

FIGURE 6. Mk 6 launcher attached to post launcher and carrying SCAR.

imately twice as strong as that used to secure the subcaliber round itself.[c]

17.3.4 Launchers for Large Aircraft Rockets

The history of development of launchers for the 11.75-in. aircraft rocket ("Tiny Tim") parallels in one respect that for the smaller forward-firing rockets—the most complicated type was investigated first and later abandoned for the simplest type. When Tiny Tim appeared on the horizon in the late spring of 1944, consideration was given to three methods of launching it: (1) displacement launchers which swing the rocket away from the airplane while holding it parallel to the line of flight and fire it automatically when it reaches the maximum separation; (2) drop launchers which release the rocket and ignite it subsequently by means of a lanyard or time delay device; and (3) fixed launchers which fire the rocket from the carrying position in the same way as for smaller rockets. It seemed fairly certain that fixed launchers of acceptable length and drag would not provide enough separation to prevent blast damage; subsequent tests showed this to be the case except for a few very rugged aircraft. The drop launcher was mechanically simple but was expected to give larger dispersions and hence required detailed study of its ballistics; thus it promised to be a relatively long-term development. The displacing gear appeared to offer the best possibility of being put into service quickly, since it promised to provide adequate separation without loss of accuracy.

The displacement launcher consists essentially of two parallel arms of equal length, with their upper ends attached to the two pivots on (or in) the airplane. A latch on the rear arm engages a lug on the rocket; to brace against side sway, latches on the front arm hold the rocket by two lugs on the for-

[c] More complete information on all of these launchers is available in many reports listed under Contract OEMsr-418 in the general bibliography in the appendix.

ward lug band. After the rocket is mounted on the launcher, it is swung up and back to the carrying position, where it is held by a standard bomb rack. In operation, the bomb rack is tripped, and the rocket swings down on the launcher arms by gravity. As the arms approach the bottom of their swing, they actuate a microswitch to ignite the rocket motor and by cam action release the lug latches. The rocket propels itself off the launcher, which is stopped in its swing by snubbing cables. The airstream throws it back to a horizontal position, where it is retained by a latch.

After certain mechanical "bugs" were worked out of them these launchers were satisfactory on rugged aircraft such as the F4U. In general, however, they had the following disadvantages: (1) major mechanical and structural upkeep problems; (2) inactivation of the bomb bay for other purposes when installed internally (as on the TBF) or (3) excessive air drag when installed externally (as on the F4U); and (4) interference with sighting because the centrifugal force of the rocket in its circular path causes a pitching of the airplane which was as much as 20 mils for lightweight fighter craft. Some thought was given to the design of a launcher with controlled displacement—one that would lower the rocket to a safe firing distance and hold it there rigidly while it was aimed and fired. None were tested, however, because of the success of the drop launcher, the next development.

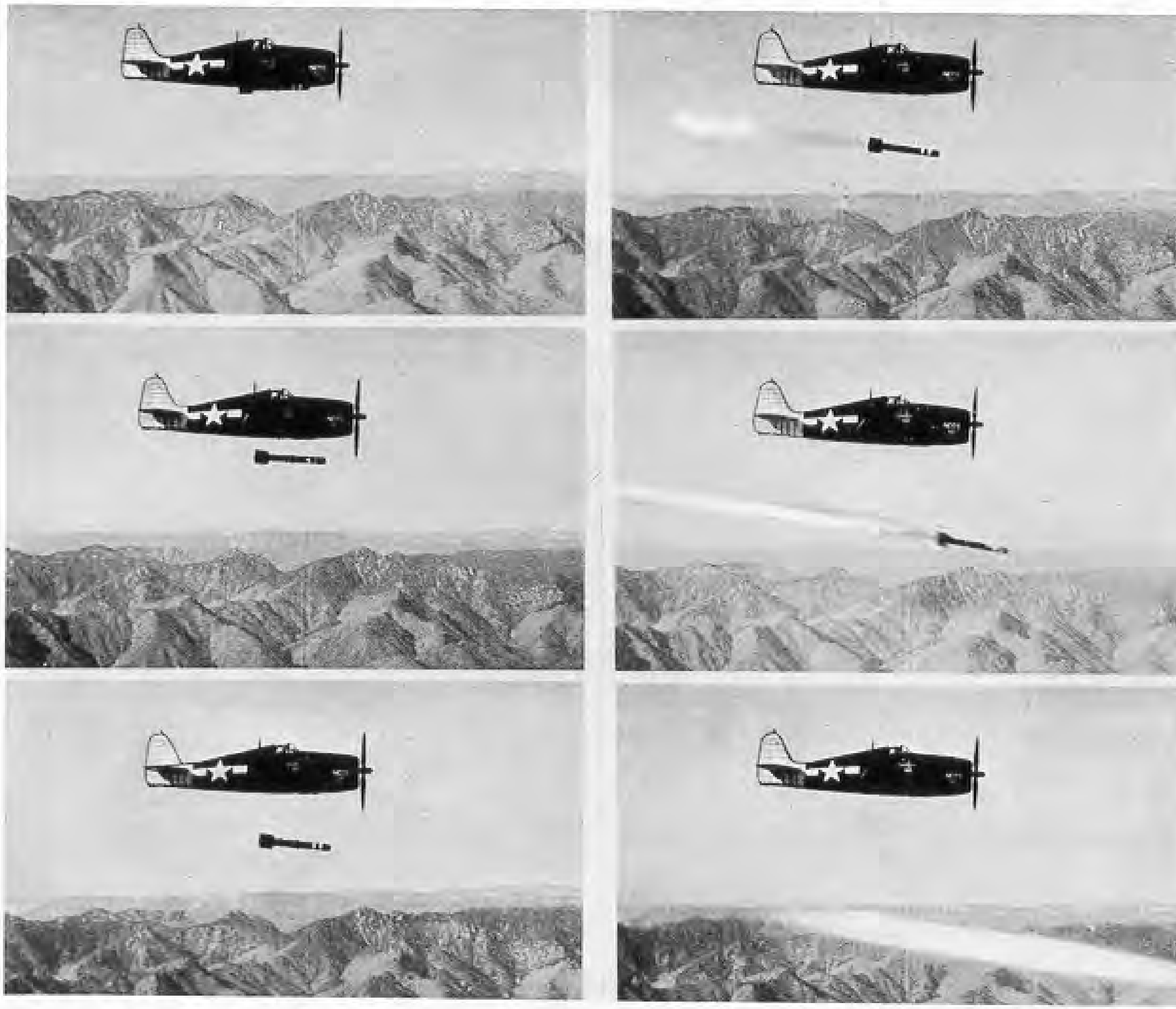

FIGURE 7. Drop launching of 11.75-in. AR from F6F.

One squadron of F4U's was equipped with displacement launchers and trained in their use, but, before it went into combat, the success of the drop launcher had made its equipment obsolete, and the squadron was recalled.

In drop launching the aircraft releases the rocket as a free body, whereupon it moves under the influence of gravitational and aerodynamic forces, connected to the airplane only by the nonrestraining cords necessary to ignite it. Ignition occurs by the closing of a lanyard switch when the cords have reeled out the appropriate separation between rocket and aircraft. Figure 7 is a series of pictures taken during drop launching of Tim from an F4U. The rounds are supported in standard bomb shackles; in some cases it is possible to use the bomb release equipment supplied with the airplane. Usually, however, special sway bracing and other supporting structure is required. Before the end of the CIT work 11.75-in. rockets had been successfully drop-launched from the following airplanes: the F6F, F4U, SB2C, F7F, P–38, and P–47 using standard bomb stations; from the A–26 using a special structure which held two rounds in the bomb bay; and from the PBJ equipped with launchers supported by a cantilever structure on both sides of the fuselage just outboard of the bomb bay. The F4U installation is shown in Figure 8.

FIGURE 8. Two Tiny Tims on pylon drop launchers of F4U-1D. Note practice heads.

During the last few months of World War II, some experimental work was done by CIT with fixed launchers for the SB2C and the P–47. The former experienced rather severe buffeting by the blast, but five pairs of rounds were air-fired from the latter with only slight disturbance to the flight. In both cases the centerline of the rocket was spaced approximately 2 ft from the wing. Because such a launcher gives smaller gravity drops and sight corrections than does the drop launcher, further work on it appears to be definitely worth while.

17.3.5 Aircraft Launcher Design Problems

In all aircraft installations the same basic problems naturally arise: attachment of launcher to airplane, attachment of round to launcher, firing circuit design, and blast problems. Their solution varied so much on different aircraft that generalization is difficult, and the weekly progress reports must be consulted for details. These problems are discussed in considerable detail in *Firing of Rockets from Aircraft*;[2] we shall mention only that of blast.

Damage to aircraft from rocket firing results from three causes: (1) ejected material from the rocket motor (closure disks, igniter wires, drying bags, etc.), (2) a shock wave from the firing of the igniter, and (3) the turbulent high-velocity airflow induced by the rocket jet. In no case was a rocket placed so that the jet itself impinged on any part of the plane (except for a few very small rockets fired from closed-breech tubes). For small rockets (i.e., 5.0-in. and smaller) the leading edges of the wings and stabilizers suffer the most damage from ejected material, and the trailing edges of flaps and ailerons are the parts most subject to damage from the combination of causes (2) and (3). Exposed fabric surfaces near the blast usually require light metal sheathing, and internal reinforcement is sometimes found necessary.

Tiny Tim, with its 150 lb of propellant and its luminous jet over 100 ft long, naturally gave blast problems of much greater severity. Anyone who is close to one when it is fired is likely to acquire a permanent feeling of amazement that such a rocket could be launched from aircraft at all. After one plane crashed immediately after firing, many weeks were consumed in tests to investigate the blast effects on various aircraft. It was shown eventually by elaborate high-speed photographic tests that the igniter shock wave was doing most of the damage, and a reduction of the igniter charge to the bare minimum consistent with good ignition removed most of the difficulty. These tests are discussed in detail in reference 9.

17.3.6 Launchers for Aircraft Spinners

The development of aircraft spinners was still in its infancy at the termination of the CIT work, and little attention had been paid to launcher designs. The launchers that were used in tests were essentially identical with the ground launchers, often attached with lug bands to the regular post launchers for finners, and probably have little similarity to the launchers which will be designed for service use to exploit the peculiar advantages of spinners.

Chapter 18

SERVICE DESIGNS OF FIN-STABILIZED ROCKETS FOR SURFACE WARFARE

By *C. W. Snyder*

18.1 INTRODUCTION

In chapters 18, 19, and 20, we shall discuss briefly each of the rockets which were developed by Project OEMsr-418 and which either were used by the Services in World War II or which had a significant influence on the design of later rockets which were used. In each case, we shall indicate the service requirements which the rocket was intended to meet and sketch the reasons which impelled the choice of particular designs to meet them. In some cases we may be able to evaluate the success of the rocket in combat, but relatively little information on this point is available and Navy or Army files must be consulted.

It is intended that Chapters 18, 19, and 20 be read in connection with the following three volumes published by CIT as part of the final report of the project: *Ballistic Data, Fin-Stabilized and Spin-Stabilized Rockets*,[1] which contains photographs, weights and dimensions, and interior and exterior ballistics data for virtually all rockets mentioned in these chapters; *Rocket Launchers for Surface Use*,[2] which contains photographs, description, and bibliography on every surface launcher which was used outside of the project itself and which the project had a hand in developing; and *Firing of Rockets from Aircraft: Launchers, Sights, Flight Tests*,[3] which, in addition to much other information, includes in the first chapter short descriptions and photographs of all service airborne launchers which CIT aided in developing.

18.2 ANTISUBMARINE ROCKETS [ASR]

The *antisubmarine rocket* [ASR] was the first American rocket to "go to war." Tests on a similar projectile began at CIT in January 1942. The actual birth of the ASR, however, was in a meeting of March 7, 1942, between representatives of Divisions C and A, NDRC. There it was decided that the projectile should be similar to the British "Spigot Gun" or "Hedgehog," except that it was to carry 40 lb of TNT in a total weight of 80 lb for a range of 200 yd. The rockets were to be fired in salvos of four or six so as to have a separation of about 20 ft on striking the water. There was sore need for such a weapon because investigation of records in Germany following World War I had disclosed that the conventional type of depth charge attack had been not nearly so effective as had been assumed, the principal reason being that sound contact with the submarine cannot be maintained at close range, and, during the interval after contact is lost but before the ship is close enough to begin its attack, effective evasive action can be taken by the submarine. Also, after the depth charges have exploded, the water is so full of echoes that it is seldom possible to regain sound contact.

Thus the requirements were as follows:

1. Range great enough so that the submarine could be attacked while maintaining sound contact with it.
2. Dispersion small enough so that a predetermined shot pattern could be laid down, calculated to give the highest probability of a hit.
3. Projectile to be capable of launching from small boats.
4. Payload great enough so that a single direct hit could inflict lethal damage on a submarine.
5. Contact fuzing so that sound contact need not be severed by an explosion unless a direct hit is scored.

The British had developed for this purpose a forward-thrown projectile called the Hedgehog from the fancied resemblance of its launcher to the animal with its spines bristled up. The launcher consisted of a group of steel rods inclined at forward angles and welded to the deck of a ship. The projectile itself looked almost exactly like the final ASR on the exterior, and its propelling tube, carrying stabilizing fins, slipped over the steel rod. Propulsion was provided by a charge of black powder in the forward end of the tube. It was desired to improve on the Hedgehog in three respects: (1) by eliminating recoil so that the projectile would be usable on small boats, (2) by increasing the payload, and (3) by increasing the range.

Twenty-three days after the meeting which authorized the rocket, firings from shipboard at sea were made. Everything about the test was satisfactory except that it was found that 85-lb projectiles were difficult to handle on the rolling, pitching, spray-drenched deck of a small ship. It was therefore decided to copy the Hedgehog almost exactly in weight and shape. Such a rocket was standardized almost immediately, and no further significant changes were made on it except the substitution of a tubular three-ridge charge for the original tubular charge with celluloid spacers. Its nose-fuzed head is 7.2 in. in diameter and 19 in. long and has only a 0.10-in. wall thickness, so that 30 of its 50-lb unfuzed weight is TNT. A later head, the Mk 5, has a still thinner wall. The 2.25-in. 11-gauge motor is equipped with a machined screw-in nozzle and a ring tail 7.0 in. in diameter. Photographs and drawings of the rocket are given in *Ballistic Data*.[1]

18.2.1 Designation and Types

In the early days when "rocket" was a restricted word, the ASR was known as the *antisubmarine bomb* [ASB], and for a time the official Navy designation was *Antisubmarine Projector Mk 20, Charge for* [ASPC]. More often than not, however, the rocket was referred to even in official communications as the "Mousetrap," the name deriving from the appearance of the launcher, which folded flat against the deck when not loaded. These designations applied loosely to any of the rockets in the ASR series.

In order to give the same velocity (175 fps) to two different heads (the one originally designed by CIT and the British Hedgehog head which was finally adopted as standard in the interest of uniformity) and three different fuzes (the HIR 1 or Mk 135, the HIR 3 or Mk 140, and the British-designed underwater-vane-arming fuze Mk 131), four different grains were designed, all having an outer diameter of 1.70 in. and a length of approximately 11.6 in., but differing in perforation diameter to give weights of 1.40, 1.43, 1.50, and 1.55 lb. Nine different combinations were originally distinguished by complete round Mark numbers, but in the latest revision of nomenclature, all ASR's are designated 7.2-in. Rocket Mk 1 Mod 0. The various combinations of components are given in *Ballistic Data*.[1]

18.2.2 Launchers

The first launcher used, and the one which gave the Mousetrap its nickname, was the 7.2-in. Type 4 launcher, designated Mk 20 by the Navy. It had four formed steel rails spread slightly apart to provide a suitable shot pattern. It could be folded flat for stowage but was not adjustable in quadrant angle since maximum range was desired. A photograph of it is included as Figure 1. Later the Navy designed a similar double-deck launcher, the Mk 22, for eight rounds. Both were used extensively, mounted in pairs on the foredecks of PC boats, Coast Guard cutters, harbor patrol vessels, destroyer escorts, and other types of vessel.

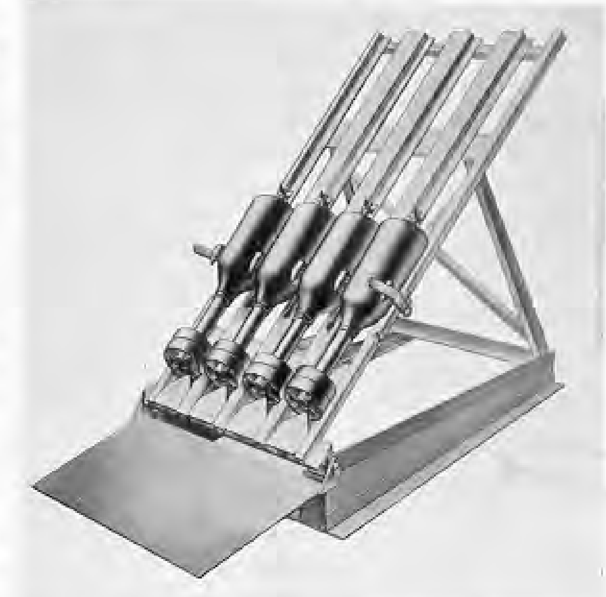

FIGURE 1. Mk 20 "Mousetrap" launcher loaded with ASR's.

18.2.3 Design Features

Nozzle. The ASR was designed hurriedly and before any extensive investigation of nozzle types had been made, and a machined threaded nozzle was chosen because it could be made easily and accurately. It is a good nozzle but is somewhat more costly than the formed nozzles that were subsequently developed.

Tail. A ring tail was chosen simply because the British had used it, but underwater tests later showed it to be a good choice. Much work on underwater ballistics of the round was done by the Morris Dam group at CIT (see Part I of this volume), and as a result of their findings two changes were made in the tail.[4,5] The four radial vanes supporting the tail rings were canted at a 10-degree angle, imparting a slow spin to the rocket and reducing "wandering" under water, and the rings themselves were streamlined, the front edge being rounded and the rear end tapered to a sharp edge, thus reducing the underwater drag and increasing the terminal velocity. The tail ring diameter was made less than the head diameter to reduce tip-off[a] which would have been significant on such a slow projectile. Since the center of mass of the whole rocket is in the head, the tail does not ride on the launcher at all.

Contacts. The system first used on the ASR motor of making electrical contact to the igniter was subsequently used on most fin-stabilized rockets. The tail shroud is composed of two rings, the rear one being welded to the radial fins and the front one being insulated. The igniter leads are provided with lugs and screwed to small metal angles inside the two rings. Spring-loaded knife contacts on the launcher make electrical connection with the rings.

General Shape. A number of alternative shapes were tested for underwater behavior by the Morris Dam group. Hemispherical noses and noses flatter in varying degrees, several tail shapes, streamlining the rear of the head, putting an air space in the rear of the head to increase the righting moment under water—all these were tested. Several of these designs gave considerably higher terminal velocities than the standard, which would be a decided advantage, but none showed any marked improvement in underwater dispersion and virtually all gave greater forward travel after impact than the standard.

Igniters. The original ASR had a brass case igniter[b] with a bakelite disk closure, and a formed celluloid "saddle" was cemented to the front end of the powder grain to hold it in place and prevent it from being squeezed between the grain and the front closure disk. In a few months the bakelite disk was superseded by a molded cellulose acetate closure, which provided a much better seal and did away with the saddle. Later the molded plastic case igniters with screw closures were specified for this rocket, as for most others.

Grains. The original grain was tubular, 11.6 in. long, 1.7 in. OD, and 0.6 in. ID. It was spaced in the tube by cellulose nitrate strips cemented to the grain with Duco household cement. It was found that strips gave 0.7 times as much impulse per pound as did the ballistite itself, and hence this fraction of their weight (and half the igniter weight) was included in the "effective weight" of the grain. Most of the experimental static-firing tests which led to the discovery of the stabilizing effect of radial holes[c] in the grain were made with ASR motors, and, as soon as the effect had been proved, radial holes were specified for all grains. The idea of extruding ballistite ridges on the grain to eliminate the necessity for celluloid strips was also tested first on the ASR and then became standard practice. Originally a cellulose acetate washer was cemented to the grid end of the grain, but it was later found to be unnecessary and abandoned.

The cast iron stool grid (Chapter 22, figure 15A) was originally specified and remained standard because the shape of the machined nozzle (Chapter 23, figure 3A) does not give sufficient port area with the box grid.

18.2.4 Reports on the ASR

The very early history of the rocket and its launcher is contained in reference 6. Further development is reported in detail in reference 7. In particular, the second volume of this report discusses the experimental tests which first demonstrated that the burning of a tubular ballistite grain could be stabilized by the use of radial holes. Instructions for use of the weapon in service are given in reference 8, and amplified and revised in reference 9. A comprehensive study of the factors determining the success of antisubmarine attacks by Hedgehog and Mousetrap projectiles is given in reference 10. See also the reports of the Morris Dam group.[11–20] Design of the grain is described in

[a] Tip-off is the reduction of the effective launching angle by gravity drop of the head while the tail is still constrained by the launcher (see Section 24.4.3).

[b] See Chapter 22, Figure 13, A and B. Igniters are discussed in Section 22.11.

[c] See Section 22.6 and reference 7.

reference 21. Reference 22 describes the two service launchers.

18.2.5 Related Rockets

As the first rocket standardized for service use, the ASR naturally inaugurated many design features which are found in later rockets. Thus the BR, the VAR series, and the SCAR (all of which are discussed in detail in the following pages) contain elements borrowed directly from the ASR. The rocket was taken over almost intact for the 7.2-in. *demolition rocket* Model 17 [DR] which was designed for the Army Engineers. It has the ASR motor and a head which is almost identical with those of the VAR series but contains the PIR base fuze Mk 146. An adapter connects the 3.25-in. head threads to the 2.25-in. motor threads. The head is filled with plastic C–2 explosive, and the rocket is intended for demolition of concrete walls and similar obstructions. Ordinarily it is used for virtually point-blank fire. Its service designations are 7.2-in. Rocket Mk 1 Mod 2 and Rocket, HE, 7.2-in., T37. It is described in reference 23.

A service launcher (T–40) was designed by Army Ordnance, although CIT assisted in its development. It consists of 20 tubes in an armored housing mounted on the turret of an M4A1 medium tank and attached to the gun so that it may be aimed by using the gun mechanism. Its predecessor, the CIT 7.2-in. Type 5 launcher, described in *Rocket Launchers for Surface Use*,[2] was superseded by the turret-mounted version because it lacked an independent train adjustment and interfered with the tank's maneuverability on rough terrain.

The DR is understood to have been used in the Normandy landings and the subsequent European campaign, but little is known about it at CIT.

Like the earlier BR, the DR was redesigned to give better accuracy by lengthening the motor and substituting a thinner-web grain. The fast-burning Model 18 has a lateral dispersion of less than 5 mils at all temperatures above 10 F when fired at 32 degrees QE from the 7.5-ft T–40 launcher. It did not get into production for service use and has no service designations.

A short-range DR was also designed for the purpose of countermining Japanese J–13 antiboat mines by firing ahead of a landing boat. This Model 19 rocket uses a standard-length motor and a 5-in. length of the thin-web grain. Fired at 45 degrees QE from the 7.5-ft launcher, its range varies with temperature from 80 to 120 ft, and its dispersion is 2 mils or less. It did not reach service use.

18.3 BARRAGE ROCKETS [BR]

The 4.5-in. *barrage rocket* [BR], originally called *beach barrage rocket* [BBR], was first suggested on June 16, 1942, just a few weeks after the standardization of the ASR, and its development proceeded rapidly. The first models were test-fired on June 24, the first full-scale sea test was on July 28, and the first service use was in the assault on Casablanca on November 8.

The requirements for the rocket were simple. No weapon existed which could fill in the gap of a few minutes between the time when the naval and air barrage had to be lifted and the time when the first invading troops hit the beachhead. This short respite from bombardment was enough to allow the enemy to organize and pour a devastating fire into the landing waves, and casualties in the first wave were alarmingly high, as everyone will remember. The rocket was intended to be carried on the troop-carrying boats themselves and to continue bombarding the beachhead up to a few seconds before the actual landing. A light-case head for maximum fragmentation and antipersonnel effect and a range of approximately 1,000 yd were suggested. Dispersion was of little importance, and in fact a relatively high dispersion might be preferable, since a large area could then be covered without the complication of having to "fan out" the launching rails.

18.3.1 Designation and Types

The original BR design incorporated a pressure-arming base fuze, but this was quickly abandoned in favor of a point-detonating fuze which detonates the head completely above ground and is thus more effective against personnel. To accommodate the base fuze, the original motors had internal threads, and, with its abandonment, the motor was simply shortened slightly leaving the internal threads so as not to have to change the head design. This was the 4.5-in. Rocket Mk 1 Mod 0, consisting of the 2.25-in. Mk 7 Mod 0 motor, the 4.5-in. Mk 1 Mod 0 head, and the original AIR fuze which was

[redacted]

a modified PDF M–52 trench mortar fuze. Shortly before the end of 1942, the design of both motor and head was changed to use external threads on the motor, and at approximately the same time an improved AIR fuze, the Mk 137, was introduced. than any other American rocket. In the latest nomenclature, all the above rockets are designated 4.5-in. Rocket Mk 1 Mod 0. The Mk 145 fuze is sometimes used in place of the Mk 137 to provide a short delay in firing; the rocket designation is then

FIGURE 2. Mk 1 Mod 0 "crate" launcher being loaded with inert 4.5-in. BR.

The new rockets had the following designations:

CIT production:

- 4.5-in. Rocket Mk 2 Mod 0;
- 2.25-in. Rocket Motor Mk 8 Mod 0;
- 4.5-in. Rocket Head Mk 2 Mod 0.

BuOrd production:

- 4.5-in. Rocket Mk 3 Mod 0;
- 2.25-in. Rocket Motor Mk 9 Mod 0;
- 4.5-in. Rocket Head Mk 3 Mod 0.

The various early design changes can be followed in the photographs and drawings of references 24, 25, and 26. Production of the Mk 3 rocket ran into the millions, and it probably saw more use Mk 1 Mod 1. A smoke head, Mk 5 or Mk 7, was designed much later, and the latest designation for the rocket with this head and the Mk 137 fuze is Mk 4 Mod 0.

18.3.2 Design Features

Motor. The nozzle, grid, grain (the 1.43-lb Mk 1), and motor closures for the first BR were the same as those of the then standard ASR, and none were ever changed, although changes in all of them were discussed at one time or another. The igniter went through the same evolution as that of the ASR as the state of the art improved.

Internal threads on the front end of the motor tube were first specified because they made the base fuze design less complicated, but, when an investigation was begun to see whether the rocket's dispersion could be decreased, evidence appeared indicating that external threads gave better dispersion, presumably because the motor pressure expanded the internal threads slightly and loosened them, increasing the malalignment. It was difficult to be certain about this point, however, and the increase in the diameter of the filling hole in the base of the head which accompanied the change from internal to external motor threads was probably a more cogent reason for the new design.

Heads. The fragmentation heads were originally made by hot pressing from standard 4.5-in. pipe, and, except for the changes in shape to accommodate changes in fuzing and in motors, they remained essentially the same. Heat treating to improve their fragmentation was soon specified. At one time there was a discussion of grooving the heads like a hand grenade, but tests showed that the fragmentation was not improved thereby. Several fragmentation tests were made,[27–29] but nothing startling was disclosed and no design changes resulted. The design of the smoke head was straightforward and involved no special problems. Several other special purpose heads were suggested and tested but never adopted.

Tails. The very first BR, which had a PIR fuze, also had a combination radial-fin tail and ring tail. The fins extended to the corners of the 4.5-in. square and were insulated from the ring, which was made in four quarters. Thus the ring formed one electrical contact, and the fins or the body of the rocket itself formed the other. This required a larger number of pieces than the ordinary ring tail, did not apparently decrease the dispersion noticeably, and somewhat complicated the launcher problem by requiring that the rocket be oriented in a certain way. It was quickly abandoned in favor of the two-ring design like the ASR, and no further changes were made on it except the simplification of making two adjacent radial fins from a single piece of metal. This became the standard design and was used on all subsequent ring tail rockets.

One other type of tail was thoroughly tested and is discussed in a report.[30] It had a single shroud ring and the insulated contact was a very short ring (about ¾ in.) inside the shroud ring at the rear. This tail was extremely simple in design and worked well, the objection to it being that it somewhat complicated the launcher contact problem. Had it been adopted, the later development of the automatic launcher would have been seriously hampered.

18.3.3 Accuracy

Although the 25- to 85-mil dispersion of the BR was adequate for its primary purpose of beach barrage, there was continual pressure to improve it so that the rocket would be better suited to other uses. For this reason, and also because the BR was a convenient test rocket for learning more about the general problem of dispersion (since it was inexpensive to make, its heads, when plaster-filled, were reusable almost indefinitely, and its dispersion was relatively sensitive to changes in motor design), a comprehensive program to improve its dispersion was undertaken and continued for several months.

The first attempt was to find a nozzle and grid combination which would give lower dispersion than the machined screw-in nozzle and the cast three-legged grid and which, incidentally, might be easier to fabricate. A considerable variety of nozzle shapes were tried along with various methods of holding the nozzles in the tube. No combination was found which gave significantly less dispersion than the standard, and some gave surprisingly large dispersions. In all cases the mechanical malalignments were known and could be corrected for, and in several tests the malalignments were made so small that they can be ignored. Despite the care with which the experiments were done, it is perhaps possible that, if they had been repeated two years later after better techniques of making formed nozzles had been developed for the aircraft rockets, the formed nozzles might have shown up more favorably in comparison with the machined. The various kinds of nozzles tried are described and the results are analyzed in references 31, 32, and 33.

Another line of attack was to try to reduce the gas malalignment[d] by straightening out the gas flow, running it through long tubes and screens and baffles of various types. None of them improved dispersion, and some made it much worse. Some

[d] For definitions of mechanical malalignment and gas malalignment see Sections 21.4.1 and 24.8.

[redacted]

experiments on spinning the BR, still using fin stabilization, were made, but the mallaunching apparently nullified the improvement in dispersion that might have been obtained. All the various expedients tried during this investigation are described in reference 34.

The final conclusion that the BR gas malalignment could not be reduced left open only one way to increase the accuracy—by decreasing the burning time. This meant either operating the motor at a considerably increased pressure or using a thinner-web grain. The use of high-strength heat-treated steel would have permitted higher operating pressures, but this was felt not to be desirable for production reasons. A safety valve for the front end of the motor was tested, which would allow the use of higher pressures over the normal operating temperature range by opening and reducing the pressure at high temperatures. As long as the valve remained closed, the dispersion was actually decreased by the smaller burning time, as expected. When the valve opened, however, the gas escaping from it had a large "gas malalignment," and the dispersion was poor. The complexity of the valve was another argument against it, and its use was never recommended. The thin-web grain was successful, however, and was recommended for service use.

18.3.4 Launchers and Service Use

The heading of this section could well serve as the title of a rather large book. *Rocket Launchers for Surface Use*[2] discusses eleven different launcher types which were designed by CIT for the 4.5-in. barrage rocket and by no means exhausts the list. A few additional launchers were designed by the Bureau of Ordnance, and a few rockets are known to have been fired in combat from makeshift wooden launchers nailed together on the spot. At least eight authorized launchers saw some service use, the most important of which were the 12-rail "crate" launcher and the "automatic."

The "crate," CIT 4.5-in. Type 5 launcher, was designated Mk 1 by the Navy, Mod 0 being for the port side and Mod 1 for the starboard. As shown in Figure 2, it consisted of twelve rails connected together into a boxlike structure and mounted on trunnions to allow adjustment of quadrant elevation from 0 to 45 degrees. The rails were 5 ft long (plus 1 in. for electrical contacts) and were considered the "standard" BR rails for purposes of measuring range, dispersion, etc.

The first service use of the crates was in the invasion of North Africa in November 1942, and thereafter they were regularly used in landing operations in the Mediterranean and European Theaters of Operations. They were introduced in the Pacific by the Second Engineer Special Brigade, which for several months had the distinction of being the only rocket unit in that part of the world.

FIGURE 3. Mk 7 "automatic" launcher loaded with various types of BR: *Top to bottom:* Mk 4 Mod 0 rocket (smoke), fast-burning motor with standard head, standard motor with incendiary head, four standard Mk 1 Mod 0 rockets.

They first used the weapon in the fighting around Finschhafen on New Guinea in October 1943, and they spearheaded their first amphibious landing at Arawe two months later. For the next six months, this group with the crates and four rocket DUKW's had a part in nearly every important landing operation. The complete story of their operations is a very long one, and it has been told by the commanding officer of the 2nd ESB, Brigadier General William F. Heavey, USA, in two articles.[35] A popularized account of the activities through 1944 is given in *The Yale Review*.[36] The crates were also extensively used on PT boats until automatic launchers, and later 5.0-in. spinners, became available, and they accounted for a large amount of enemy shipping.

The first automatic launcher (Figure 3) was made at CIT in April 1943, and its important advantages were immediately recognized:

1. Light weight. For twelve rounds, its weight is only 115 lb as compared to 350 for the crate.

2. Simplicity. On electrical wiring, for example, the automatic requires only one-twelfth as much as other launchers, since a single set of contacts fires the whole salvo.

3. Greater safety for the operator, since it can be loaded from the side, whereas most other multiple launchers require either breech or muzzle loading.

4. Adaptability to a great variety of installations. Its almost universal adaptability is the launcher's primary advantage. In *Rocket Launchers for Surface Use*,[2] seventeen typical installations are listed, and how many more actually existed is probably impossible to determine. With this launcher, an LCM–3, for example, could fire a ripple salvo of 576 rockets—a total of 12,000 lb of payload laid on the target in about 4 seconds if desired. Even the lowly jeep carried heavy artillery as shown in Figure 4.

FIGURE 4. Mk 7 launcher installation on jeep.

The automatic, designated Type 8 by CIT, became the Navy Mk 7 launcher, and production by the Services ran into tens of thousands. It is probably no exaggeration to say that the 4.5-in. barrage rocket was the most important rocket used in World War II and the Mk 7 was the most important launcher. It was this combination that helped to teach the Japanese that they could not defend a beach and resulted in virtually no opposition being offered to the initial wave in the landings during the last year and a half of World War II.

[redacted]

18.3.5 Reports

There are a large number of reports on the procedure for using the BR with various launchers, and for these the reader is referred to the bibliographies accompanying the descriptions of the launchers in *Rocket Launchers for Surface Use*.[2] References 24, 25, and 26, although also intended as service manuals, include sufficiently complete descriptions of the rocket as it was at the time of their writing to give a picture of the various steps in its development. Photographs of the very earliest round and the original crate launcher may be found in reference 37. Manufacturing methods used in CIT pilot production are described in reference 38. Tests made in developing the fast-burning grain are discussed in reference 39. Some tests carried out to learn about the suitability of the BR for paratroopers' use are described in references 40 and 41.

18.3.6 Related Rockets

The CIT answer to requests for better accuracy with the BR was the so-called "fast-burning" or "short-burning" BR. Very little development work was required on this round. The perforation in the grain was simply enlarged to give a web thickness of 0.4 in. instead of 0.55 in., and grain and motor tube were lengthened to bring the propellant weight back up to 1.43 lb. The long thin grain gave a rather severe drop in effective gas velocity [e] at high temperatures, but the reduction in range was only 20 yd at 115 F, and the upper temperature limit was sufficiently high. The comparison of this rocket and the standard with regard to lateral dispersion at 45 degrees QE is shown in Table 1.

TABLE 1

Temperature (degrees)	Burning time (seconds)		Lateral dispersion (mils)	
	Standard	Fast-burning	Standard	Fast-burning
10	0.66	0.38	85	48
70	0.37	0.22	45	20
120	0.23	0.14	25	4

This rocket was recommended for service use but was never adopted because, by the time it was ready, production on the standard model was well under way and the rockets were needed so urgently that it was not thought desirable to introduce the change. The principal production difficulty would have been to change the fuze, which was the critical item. The much shorter burning time of the new model required extensive modifications of the AIR fuze.

A 250-yd barrage rocket was also developed for possible use in detonating land mines. The thin-web charge was used to keep the burning time short, since a dispersion of 5 mils or less was desired. Standard length motors were used even though the grain was less than half the standard length, because tests showed that the rocket had insufficient stability with a shorter motor and gave bad dispersion. The combination of short burning time and low velocity gave a relatively low dispersion, less than 8 mils at medium and high temperatures, but still considerably above the desired value. The request for this rocket was withdrawn before development work was entirely complete.

18.4 CHEMICAL WARFARE ROCKETS [CWR]

Development of a rocket for the Army Chemical Warfare Service was one of the earliest projects tackled by the CIT group, the first field firing being on December 23, 1941. The intention was to develop a rocket to replace or supplement the Livens projector bomb, since the lack of recoil would permit the launcher to be mounted on a truck, eliminating both the weight of the Livens mortar and the time required to emplace it. The original specifications called for a projectile to carry a liquid payload of between 20 and 30 lb with a maximum range in excess of 3,000 yd. No definite specifications as to dispersion were made, but it was indicated that a dispersion of the same order as that of the Livens (probable error 50 yd in range and 25 yd in deflection) would be acceptable. Following the first tests of the projectile at Edgewood Arsenal, Maryland, more definite specifications were outlined, calling for a bomb of 2.2-gal capacity to carry 20 lb of chemical agent for a maximum range of 3,400 yd or more.

The first rocket designed had a motor which was patterned closely after the British 3.25-in. motor, in that it had a formed nozzle of almost identical shape, sealed at the front end by an obturator cup and held in the motor tube by a piston ring at the

[e] See Section 21.1.1.

[redacted]

rear (the British RP-3 uses rivets), a six-legged stool grid seated on a nozzle ring, and four radial fins. The motor was 3 ft long and extended clear through the center of the 7.0-in. diameter head to make a compact, streamlined-looking projectile. Because of its short length, the rocket had a dispersion of more than 100 mils and was unacceptable also because the re-entrant design precluded the use of a burster tube in the head to disperse the contents upon impact. Moving the motor back so that it was completely behind the head reduced the dispersion to about 40 mils and necessitated adding a 7.0-in. ring to the tail to support it on the launcher. The ring was made in two parts to provide electrical contact as with the ASR and BR, but the radial fins were left unchanged, extending forward and radially beyond the ring. Tests with inertia-firing fuzes (both nose and base) indicated that a point-detonating fuze was required to give sufficiently rapid dispersal of the load. Except for the lack of a suitable fuze, the rocket was regarded as satisfactory for the contemplated service use.

FIGURE 5. Launcher, Rocket, Multiple Artillery, 7.2-in., T32, mounted in truck and loaded with 24 CWR-N's.

Two changes in design were then made: substituting a box grid for the complicated cast stool grid, an improvement which became permanent, and forming the nozzle in one piece with the tube instead of inserting it. It was found that the one-piece nozzles deflected considerably under hydrostatic pressure,[42] but no clear evidence of increased dispersion from this source was discovered.[f]

On May 29, 1943, a meeting of the JNW Rocket Board decided, in the interest of uniformity of design and interchangeability of auxiliary equipment, to increase the head diameter from 7.0 to 7.2 in. and to use the Mk 3 VAR (retro rocket) motor (see Chapter 19) with a different grain for the CWR. The new model was dubbed the CWR–N (N for "new"), and it showed a lateral dispersion of approximately 66 mils as compared to 40 for the old model. Radial fins were therefore added to the ring tail, and the dispersion returned to its former value (see Figure 11B of Chapter 23).

[f] See reference 43 for comparison of dispersion of various models of CWR.

[redacted]

18.4.1 Design Features

Grain. As was the case with all early rocket motors, much trouble with the grain was experienced in the beginning, and a number of things were tried until static-firing tests on the ASR solved the problem. A 2.5 x 1.0-in. tubular grain with 24 radial holes was then adopted, and, except for the addition of three ridges, this remained the standard.

Fuze. The Mk 137 BR fuze was used on the CWR for a time, but the large propeller was not necessary for such a fast projectile and the large protective cup surrounding it reduced the range by 100 yd. The Mk 147 fuze with a much smaller propeller was developed especially for the CWR. In accordance with standard practice for chemical bombs, the fuze detonates a burster tube which extends virtually the full length of the head.

Nozzles and Accuracy. Although satisfactory in other respects, the CWR suffered from the usual ailment of rockets—insufficient accuracy. As previously mentioned, the accuracy was improved on two occasions by tail changes which increased the stability. At several times during the long period of development of the CWR, tests of various nozzle changes were made in an attempt to decrease dispersion. The first nozzle had an abrupt entrance cone similar in contour to that of the machined ASR nozzle (see Figure 3 of Chapter 23), and this was changed to a more gradually tapering entrance on the basis of yaw machine tests which showed that longer nozzles gave smaller side forces. It is probably impossible to draw any conclusions from the data on the effect of the later change from insert to integral nozzles, because the observed dispersions varied so widely from test to test and various other factors were being changed from time to time. It was thought that the reduction in nozzle expansion ratio entailed in the change to the VAR–type motor might increase dispersion, but the field tests of this point gave negative results.[44] The difference in dispersion between rockets having nozzles with smooth and rough interior finishes was found to be so small that it could not be clearly separated from the malalignment effect. It was thought that the orientation of the nozzle throat, or more accurately of the portions of the entrance and exit cones close to the throat, might be more important than the orientation of the exit cone which was normally assumed to define the direction of gas flow. Tests indicated that "throat malalignment" does have an influence on dispersion but that it is less important than the ordinary "mechanical malalignment."[45] The only change in nozzle design which ever gave a spectacular increase in accuracy was the elimination of rough and irregular welding at the nozzle exit circle.[44]

Other factors which were investigated for possible effect on accuracy were oscillation of the liquid filler in the heads, variation in filler density, and launchers with varying lengths of overhead guides. On none of these tests were any definite positive results obtained.

18.4.2 Designation and Types

The older models, called CWB or CWR in CIT reports, had no service designations. The CWR–N motor is designated 3.25-in. Rocket Motor Mk 5 Mod 0. Two 7.2-in. heads have been standardized: the Mk 7 for chemical fillers and the Mk 9 for TNT, the latter being nearly 3 in. shorter to accommodate the higher-density filler without increasing overall weight. Complete round designations are "Rocket, Chemical, 7.2-in., T21" and "Rocket, HE, 7.2-in., T24."

18.4.3 Launchers and Service Use

The standard CWR launcher is the CIT 7.2-in. Type 2, designated by the Army as "Launcher, Rocket, Multiple Artillery, 7.2-in., T32." It is a 24-rail launcher 10 ft long, very similar in design to the BR crate, which can be mounted on the ground or in the bed of a 2½-ton truck as shown in Figure 5. Although Army Ordnance produced a considerable quantity of the launchers, no service use of the CWR is known.

18.4.4 Reports

The early development of the CWR is recounted in detail in reference 46. Propellant development is discussed in reference 47. See also reference 70 for a report on high-speed water tunnel tests.

18.5 TARGET ROCKETS

The development of rockets as targets for antiaircraft training antedates the CIT contract. It was undertaken jointly by Sections H and E, Division A, NDRC, in August 1941, and three flight tests were conducted in the East before OEMsr-418 was organized. In the summer of 1941, when NDRC had circularized the Armed Services as to their interest in a variety of proposed rocket projectiles and devices, the Coast Artillery, which had the responsibility for training antiaircraft gunners, answered that they would like to have a target rocket developed. Something was needed which would give gunners adequate practice in firing at targets which approximated the speed and courses of aircraft. Small radio-controlled airplanes or drones would have served the purpose, but they were not in quantity production and were neither cheap enough nor fast enough. The conventional towed sleeve target was too slow and moved on too steady a course to give the necessary training in "leading" a high-speed, maneuvering target. Target rockets could be fired toward, away from, or across the line of fire of the guns either in low straight paths or in a high looping trajectory, and should be able to simulate most of the situations met in battle.

The newly formed CIT group took over the development begun in the East. There were several advantages in beginning with the target rocket. It was a less complicated problem than most others, since neither head nor fuze were required. The main requisites were velocity and visibility—a motor with sufficient thrust and fins of sufficient size. The experience in designing the motor and firing the target rockets would give useful data which could be applied to the more difficult problems which were being undertaken while the target rocket development was proceeding.

The work was handicapped by troubles with propellant—both quantity and quality, and the early history of the target rocket is the history of the development of satisfactory propellant.[48] Rockets were made, however, even though for a few months propellant failures were rather frequent, and on November 29, 1941, the 78th Coast Artillery at the Mojave Antiaircraft Artillery Range got a chance to shoot at three target rockets. This was a small beginning, but "rocket shoots" rapidly became bigger and more frequent. The verdict of officers and men was uniformly favorable. By the summer of 1943, for example, the target rocket range at Camp Pendleton, on the southern California coast, was scheduled for four weeks in advance. By the following December, when the Bureau of Ordnance standardized and undertook the production of two types of rockets and several launchers, the CIT group, developing and producing its own target rockets, had participated in training some 21,000 men. Improvements in the rocket design continued to be made up until that time.

18.5.1 Design Features

Motor. The first CIT target rocket motors were very similar to those which had been tested in the East, 3.25 in. in diameter and approximately 6 ft long. The only propellant available was 1.0 x 0.25- or 0.87 x 0.25-in. tubes 5 in. long, and they were strung on a steel "cage" attached at the front end of the motor. In one design, the whole rocket was motor, and the gas from the propellant charge at the front had to traverse a long empty space to reach the nozzle. In another design, the rocket was jointed in the middle, the rear part being motor and the front part empty tube. The latter design was tried out because it was thought that smaller heat losses might give better efficiency. Tests showed that the long motor with the dead space gave slightly smaller gas velocities, longer burning times, lower average pressures, and considerably smaller differentials between peak and average pressure. It was chosen as standard primarily because it was cheaper and easier to produce. Fourteen-gauge tubing (0.083-in. wall) was used for the motor in order to save weight, and the pressure was kept low by using nozzle K's around 180 or lower (see Section 22.4).

Earliest models had machined screw-in nozzles, but the spun integral nozzle (see Figure 4A of Chapter 23) soon became standard. A front closure involving an obturator cup and a piston ring was worked out, so that no threading was required on the motor tube. For attaching the fins, L-shaped or T-shaped lugs were welded to the tube.

When single large grains became available, the propellant was moved back to the rear of the motor, and the grid was seated on a grid ring which was held in place by welding through four holes drilled through the tubing.

After set-back fins were developed, the motor was simply shortened to 32 in. without other modification. (See Figure 6.)

Tests of multinozzle motors with the set-back fins were made, and three-nozzle motors were found to be satisfactory provided that the nozzles were

[redacted]

aligned with the fin longerons. Random orientation of the nozzles with respect to the longerons apparently increased the dispersion.

For its production, the Bureau of Ordnance preferred a motor design similar to that of other motors in production. Hence 11-gauge tubing was

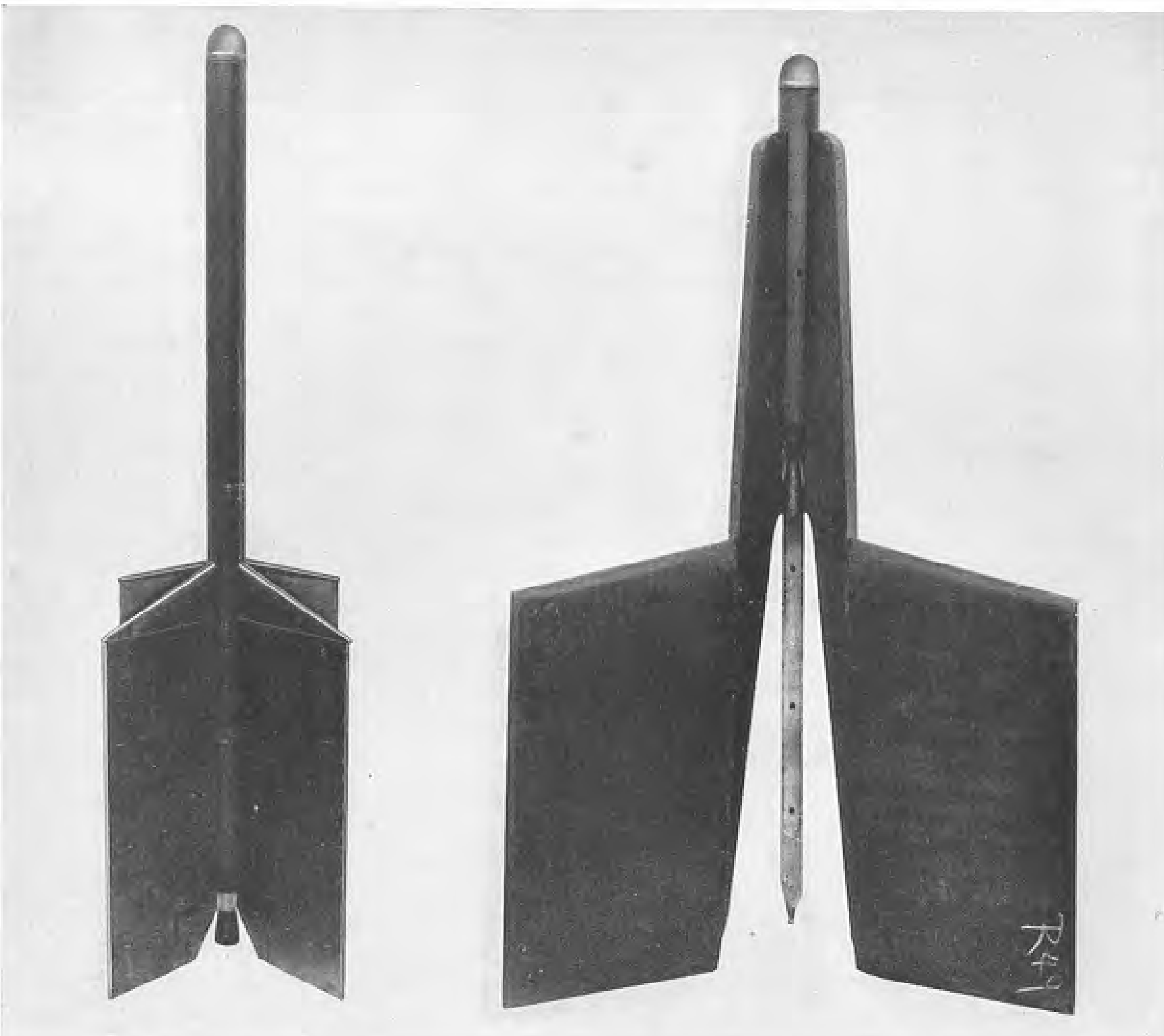

FIGURE 6. CIT target rockets. *Left:* eary 4-fin variety. *Right:* set-back fin variety with straight fins (final design has fins canted slightly to impart slow rotation).

specified and a nozzle design like that of the Mk 7 AR motor (see Figure 4C of Chapter 23). Stud-welded bolts with washers and nuts were also specified in place of the L-shaped lugs, and a threaded front closure was substituted for the piston ring. These changes increased the weight and decreased the velocity somewhat, since the propellant grains were not changed.

Propellant problems for the target rocket were straightforward once the technique of making good grains had been learned, and no special comment on them is required.

Fins. The fin problem for the target rocket is obviously entirely different from that for any other rocket. In addition to their being as large as possible, the chief requirements are that they be relatively cheap to manufacture and able to withstand the weather. As with any rocket part, it is desirable also that they be light. A large variety of fin constructions was tried, and it would serve no purpose to detail them here. In all cases they were made relatively thick, with an internal structure to withstand the acceleration and wind forces, and were covered with some thin material; aluminum, burlap, cloth of various types, plastic, and paper were among the things tried.

The 6-ft motors had 4 fins approximately 1 ft wide and 3 ft long. After static firing had showed that the hot part of the jet does not spread very rapidly on leaving the nozzle, tests began on target rockets having the fins extending back beyond the nozzle exit plane. It was found possible by attaching them to tapered longerons to set the entire fin back of the nozzle and thus do away with the necessity for a long motor which had been required to move the center of mass forward. Improved fin construction together with the much smaller accelerations given by the thick-web single-grain charges made possible an increase in fin width to 18 in. With the wider fin, the number could be reduced from 4 to 3 and still increase the visible area over that of the long 4-fin target.

The fin construction which was finally worked out as the most satisfactory was to cover the framework with $\frac{1}{16}$-in. fiberboard attached with staples and glue, and then to spray the whole fin with a coating of hot paraffin to make it watertight. The framework is made of high-quality white pine with tongue-and-groove joints nailed and glued. Such fins could be manufactured for less than $7 each in small quantities.

It was found that canting the fins slightly gave a considerable decrease in dispersion, and with the set-back design it is easy to do. The longerons are fastened parallel to the motor axis and the fins are attached to them with the rear end displaced $\frac{3}{8}$ in. from the front end. The resulting rotational velocity is between 1 and 2 rps.

Contacts. Electrical contacts on a molded bakelite cap which sealed the nozzle were used for a time. Since rapid loading is less important than certain contact, they were later replaced by ordinary household-type electric plugs which fit into receptacles on the launchers.

18.5.2 Launchers

The older type target rockets were fired from a simple launcher consisting of two pieces of $1\frac{1}{2}$-in. tubing with one fin extending down between them. With the standard design, this system was not practicable, and a so-called "M-rail" was designed which contacts the motor tube and two adjacent longerons. It can be mounted on the standard M1 trailer launcher for Army target rockets. Separate trailer mounts and tripod mounts have also been used.[49]

18.5.3 Designations and Types

Target rockets of various velocities from 450 mph down to 200 mph were used at various times. They were usually differentiated in CIT reports by drawing number, 3T4, 3T7, etc. Two velocities were finally chosen, 290 mph and 415 mph, to be used respectively for beginning and advanced training. In the final designation, all fast rockets are called 3.25-in. Rocket Target Mk 1 Mod 0, and all slow ones are 3.25-in. Rocket Target Mk 2 Mod 0, except that in either case the addition of a flare for night firing changes the Mod number to 1. The earlier designation distinguished between CIT (Mk 1 fast, Mk 2 slow) and Bureau of Ordnance (Mk 3 fast, Mk 4 slow) production.

18.5.4 Reports

Summary reports which discuss the development of the rockets themselves as well as the training tactics are references 50, 51, 52, and 53. Reports on methods of training, scoring equipment, etc., include references 54, 55, and 56. On manufacturing problems, see reference 57.

18.6 LITTLE ROCKETS

The various rocket projectiles with 1.25-in. motors developed by CIT are so closely related that they will be discussed together. They will be treated somewhat more briefly than the larger rockets because they are less important from the standpoint of service use and also because, once the fundamental principles of rocket design had become understood, their development was relatively straightforward and simple. They did have an important place, however, both in providing information necessary for the design of larger rockets and in training Service personnel in their use.

18.6.1 Chemical Warfare Grenade [CWG]

The *chemical warfare grenade* [CWG] was one of the earliest CIT projects, the first model having been fired in the desert test of December 23, 1941, which saw also the introduction of its big brother the CWR. The specifications laid down by the Chemical Warfare Service were payload, 1 lb of liquid in a frangible container; range, 600 yd; accuracy, 5 mils; trajectory, not to exceed 30-ft height in 200-yd range; projector, to be carried and operated by one man. Since it was thought of as primarily for use against tanks, the rocket was originally called the *antitank grenade* [ATG], but this name became obsolete within a month.

The service history of the CWG was disappointing. When it was demonstrated to the Chemical Warfare Service in April, the reception was unenthusiastic because the bakelite head could not be used for FS, although the observers were pleased with its accuracy. Later tests showed that 1 lb of liquid was not sufficient to cause the damage required. Research on the CWG was therefore stopped, but it had served an important function in making possible a large number of experimental tests with little expenditure of propellant, which was extremely scarce, and had yielded much information.

Motor Design Features

The first motors were necessarily designed to use the only solventless-extruded ballistite tube then available ($\frac{15}{16}$ in. OD by $\frac{1}{4}$ in. ID) since tests had already demonstrated the marked superiority of this material to solvent-extruded tube. They were made of 1.25-in. outside diameter, 16-gauge steel tubing, threaded on the outside to take the front closure and on the inside to take the machined nozzle. Several motors burst at the undercut of the nozzle thread when the motor became hot, and the screw-in nozzle design was therefore abandoned in favor of the spun integral design. Several hundred rounds of CWG motors were fired on the yaw machine, and the following facts were learned:

1. Longer nozzles tended to give smaller side forces (range firings appeared to corroborate this with smaller dispersion).
2. Integral nozzles were frequently distorted and cocked out of line by the heat and pressure of firing.
3. If the burning time were short enough, the distortion did not occur.
4. Side forces could be reduced by careful alignment of the nozzle exit cone.

Also shortly after the first CWG firing, the first calculations of malalignment effect were made, which showed that accuracy could be considerably improved by decreasing the burning time.

On the basis of all this information, a double-web charge was designed to reduce the burning time as much as possible. It consisted of a $\frac{1}{2} \times \frac{1}{4}$-in. tube inside a $1.0 \times \frac{3}{4}$-in. tube and gave a burning time of only 0.12 second at 70 F. Rounds were checked on a malalignment balance and carefully straightened to keep their malalignments small. The result was that the dispersion dropped from its original very large value to approximately the 5 mils desired. Experiments continued to attempt to reduce the dispersion of the single-web charge, but they were doomed to failure by the gas malalignment, which at that time, of course, was not understood. A large number of field tests of the CWG with various launcher lengths, various burning times, and various fin sizes during 1942 established the correctness of the theory of dispersion which was published in reference 58 and formed the basis for all subsequent finner development.

To avoid the complicated assembly operations involved with the tubular double-web charge with its numerous celluloid strips, a 1-in. 4-spoke or "okra" grain was extruded. Since its burning time was short, it gave good dispersion. Its gas velocity was somewhat smaller because of the slivers left at the end of burning, and considerable difficulty was found in trying to make the cylindrical portion and the spokes burn at the same rate. Interest in this grain shape declined with the abandonment of the CWG.

The grid originally used for the tubular double-web charge was a complicated structure of four legs and a ring. An adaptation of the box grid having six pieces instead of four was found to work better for both double-web and 4-spoke and was adopted as the final standard.

The double-web CWG is shown in Figure 7. Features other than the motor require little comment. The head was formed of an 18-in. bakelite tube, 1.62 in. in outside diameter, closed at the front end by a hemispherical steel cap and extending back several inches over the motor to a supporting shoulder of sufficient width to maintain accurate alignment. Several types of stabilizing fins were tested, including a folding design with four 4 x 1.12-in. blades which opened by means of springs from a rectangle 1.62 in. square to a radial position. They worked satisfactorily, but were abandoned as too complicated, and four fixed radial fins were adopted.

Reports on the CWG

See references 58, 59, 60, 61, and 62.

use the same head as the 200-fps, but the very long grains gave such low gas velocities that the required velocity was not reached. Rather than extrude a thicker-web grain which could have given enough impulse, it was decided to retain the 1.0 x 0.5-in. shape and use another still lighter head.

The heads are identical except for length and are machined from 2.5-in. steel bar stock. Originally they contained a shotgun shell which, on contact with the target, was set off by a firing pin which projected in front of the head. The shells were found to be unsafe to handle and unnecessary since the thud of the head itself on the submarine hull was enough to tell the occupants that a hit had been scored. The shell was therefore abandoned.

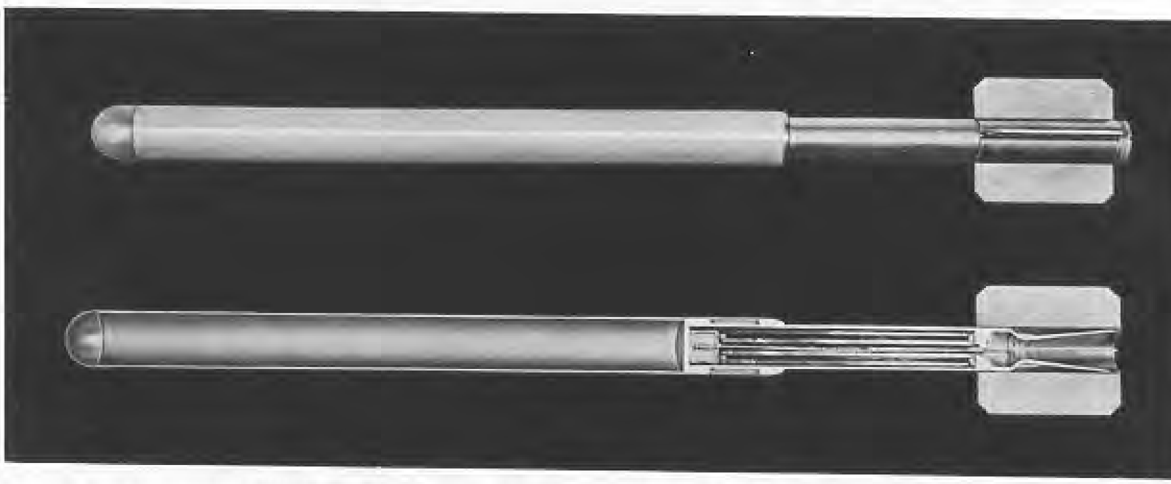

Figure 7. Assembled CWG and cutaway showing nozzle and head construction and double-web tubular charge.

19.6.2 Subcaliber Rockets

The acute shortage of ballistite which plagued us constantly during the first two years of World War II made the development of subcaliber practice rounds imperative, although they would have been useful in any case. Development of the subcaliber ASR (175 fps) began almost simultaneously with that of the drift signal rockets (see Section 19.1.6), and by adjusting the head weight it was possible to use the fast drift signal motor with the addition of a ring tail for the practice ASR.

The increase in velocity to match the 200-fps VAR was accomplished without changing the motor simply by shortening and lightening the head. It was hoped that the 300-fps subcaliber VAR could

Launchers for the subcaliber rounds are boxlike, formed by bending steel sheet, and adapted to be attached to the launchers for the full-scale rounds. The one for aircraft firing (2.5-in. Rocket Launcher Subcaliber Mk 3 Mod 0) is shown in Figure 8.

Reports

The only official CIT report on the subcaliber ASR projectile itself is reference 63, which was written very early in its development when it still used the CWG motor. The subcaliber VAR's are discussed in several of the reports on full-scale ammunition (see Section 19.1.4). Reports by the Morris Dam group on the underwater performance of the various models include references 64, 65, 66, 67, 68, and 69.

[redacted]

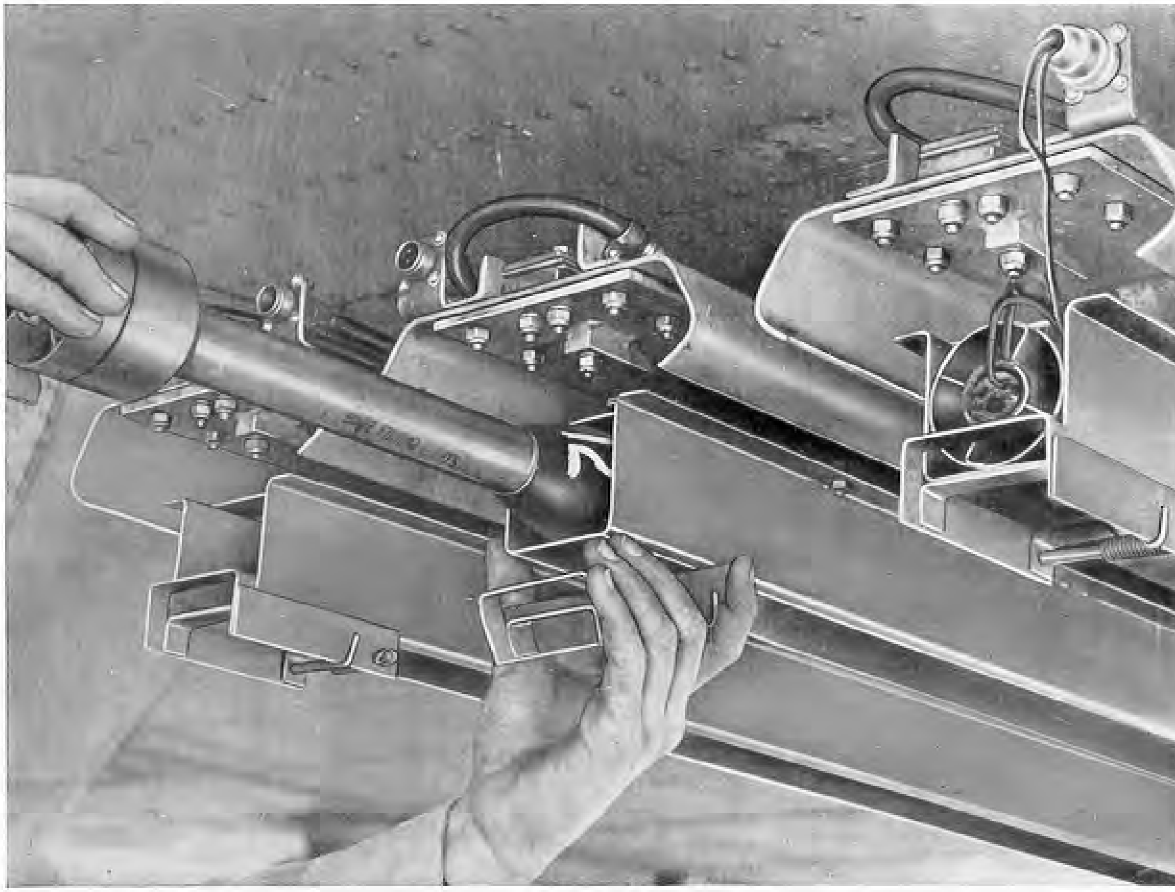

FIGURE 8. Subcaliber VAR being loaded into 2.5-in. Launcher Mk 3 mounted on 7.2-in. Launcher Mk 1 Mod 1 under wing of PBY-5. Compare with Figure 1 of Chapter 17.

18.6.3 Rocket Grenade

The last rocket having a 1.25-in. motor which was developed was the *incendiary rocket grenade* [IRG]. Its purpose was very similar to that of the CWG, which it resembled also in its history, since it was tested extensively and brought to completion at just about the time that the Services decided that they had no use for it. Like the CWG it was to be fired at a low angle and required high accuracy, but its head was provided with a very simple shear-wire impact fuze to disperse the contents rather than relying simply on the shattering of the head on impact.

The motor was essentially the same as 1.25-in. Mks 1 and 2 except that to increase the accuracy a nozzle with a long entrance cone and a smaller throat was used. The long nozzle gave slightly less gas malalignment in addition to lengthening the round. The smaller nozzle gave a pressure of about 2,000 psi at 70 F instead of 1,000 psi for the subcaliber motors, thus reducing the burning time and the dispersion. Approximately 5-mil accuracy was attained at medium temperatures with a launcher only 3.5 ft long.

Since the rocket was intended to be fired from a back-pack launcher which could be worn by an infantryman, several rather unorthodox launcher designs were tried, but development was not completed.

[redacted]

Chapter 19

SERVICE DESIGNS OF FIN-STABILIZED ROCKETS FOR AIRCRAFT ARMAMENT

By *C. W. Snyder*

19.1 RETRO ROCKETS [VAR]

HARDLY HAD THE antisubmarine rocket [ASR] program (see Section 18.2) gotten under way in the summer of 1942 when high-priority experimental work began on the problem of adapting the rocket to aircraft use. The development of the *magnetic airborne detector* [MAD] had made it possible to detect a submerged submarine directly beneath the airplane, but, by the time the target was detected, it was already too late to use the conventional type of bomb. It was suggested that by rocket propulsion a bomb could be given a velocity equal and opposite to that of the aircraft so that it would fall almost vertically from the point of firing and hence could be triggered by the signal from the MAD.

The first question to be settled was which direction to point the rocket. During burning while the rocket is picking up speed, its velocity relative to the ground is less than that of the airplane so that it is moving in the same direction as the airplane but with decreasing velocity. (See typical trajectory in Figure 1.) If the rocket is accelerated by its own motor, it will be moving backwards through the air during the whole burning time, and in this attitude the tail fins will increase the yaw instead of decreasing it. It was thought, therefore, that the rocket's flight might be better if it were pointed in the direction of the airplane's motion and pushed out of the launcher by an auxiliary rocket, called a "mule," which separated from it after the end of burning.

The first firing of an American rocket from aircraft in flight took place on July 3, 1942, when several ASR's were fired backwards from a PBY-5A. The "mule" tests were made with a nonstandard type of ASR with a round nose and a streamlined afterbody, but, for the tests in which the ASR was accelerated by its own motor, standard ammunition was used. It was quickly found that the latter system gave only about one-third the dispersion of the former, so the use of the "mule" was abandoned.

19.1.1 Designation and Types

The ASR was satisfactory for vertical bombing from the PBY, but it was too slow for most other airplanes on which installations were contemplated, and a new series of motors had to be designed. Initial tests were carried out with 2.75-in. motors, and a tubular three-ridge grain suitable for this caliber was extruded and tested. The development was not completed, however, and 3.25-in. motors were chosen for the purpose because (1) the grain which could be put into a 2.75-in. motor would not give the 400-fps velocity required for the B–24 airplane without unduly increasing the burning time, and (2) 3.25-in. motors had already been developed for other purposes (it was in fact one of the first sizes on which work had been done by the project) and it was felt desirable to keep the number of different motor calibers to a minimum in the interest of standardization and simplicity. This decision was made in December 1942, and by May 1943, three 3.25-in. motors had been developed and standardized.

The terms "retro" and "vertical" have been used rather loosely and usually interchangeably to describe rockets fired backwards from aircraft, although it was originally suggested [1] that "vertical" be used to describe bombing in which the rocket's velocity simply canceled that of the aircraft and "retro" be reserved to apply to the case where the rocket has considerably more velocity than the aircraft so that its fall relative to the earth is no longer approximately vertical. Since the original intention was to use truly vertical rather than retro bombing, the series of rockets designed for this purpose were known as *vertical antisubmarine bombs* [VAB or VASB] and later as *vertical antisubmarine rockets* [VAR] (see Figure 2). In cases where the Mark numbers (of which there are two sets as for the ASR) are not given, the members of the VAR series are usually distinguished by their velocity or by their drawing numbers: 7V11, 7V12, and 7V13.

The three VAR motors differ from each other

[redacted]

only in the following respects: (1) motor length, (2) grain (originally it was intended to have the grains differ only in length, but it turned out to be preferable to use a slightly thinner web for the shortest one), (3) nozzle diameter and contour, and (4) length of igniter leads. The motors are designated:

3.25-in. Rocket Motor Mk 1 Mod 0 (7V11, 210 fps);
3.25-in. Rocket Motor Mk 2 Mod 0 (7V12, 310 fps);
3.25-in. Rocket Motor Mk 3 Mod 0 (7V13, 400 fps).

All use the Torpex-filled Mk 6 head. Complete round designations, old and new, are given in *Ballistic Data*,[2] which also lists several other combinations.

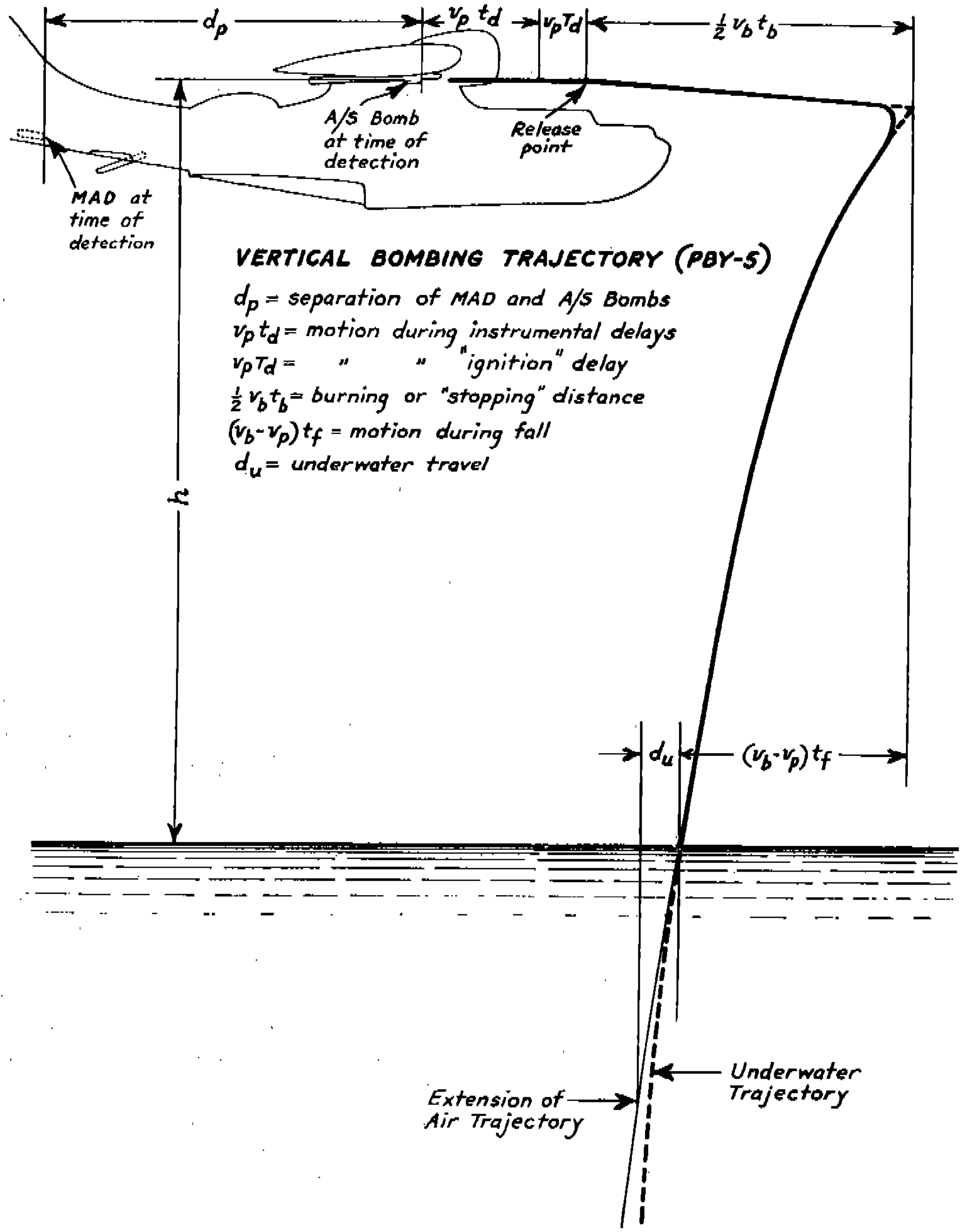

FIGURE 1. Typical vertical-bombing trajectory.

[redacted]

19.1.2 Design Features

Heads. The Mk 6 head is patterned after that of the ASR but is slightly shorter and has a thinner wall.

Fuzes. For submerged submarines, the ASR fuzes—either HIR or underwater-vane-arming—worked satisfactorily in vertical bombing, but it was felt desirable to have a fuze which would function also against surfaced subs, and development of such a fuze was undertaken. Considerable work was done with AIR-type fuzes, but the velocity of the rockets relative to the air was so low during most of their flight time and their yaws were so large that it was difficult to make a propeller work reliably. The solution was the SIR (see Chapter 16) which was armed after a specified number of rotations of a flywheel by a clockspring. It was designated the Mk 139 Mod 0 and was used on all vertical bombing rockets except the ASR.

Motors. The relatively low stability of the *barrage rocket* [BR] with its 2.25-in. motor and 4.5-in. tail had indicated that a considerable decrease in tail efficiency could be expected with the 7.0-in. tail if the motor tube diameter was made too large. Firings of the 2.75-in. motors gave good results, but those with 3.25-in. motors indicated decreased stability. Hence it was decided to reduce the tube diameter to 2.75 in. at the rear, as shown in Figure 4D of Chapter 23, in order to get more air through the tail. Formed insert nozzles were chosen for cheapness since accuracy was no problem, the dispersion of the rockets in vertical bombing being less than the uncertainty in the position of the aircraft. Because of the swaged tube, it is necessary to insert the nozzle from the front end, and to obviate having to press the nozzle in the whole distance, thus galling the inside of the tube, the tube's inside diameter is reduced from 3.01 to 2.9 in. in the region where the front end of the nozzle and the grid are located. This reduction in port area increases the internal K of the motor, but with low-performance motors it is not serious. Box grids are used.

Tails. The tail design is identical with that of the final ASR, but the shroud rings are 7.2 in. in diameter instead of 7.0, so that the same lug band will fit both tail and head.

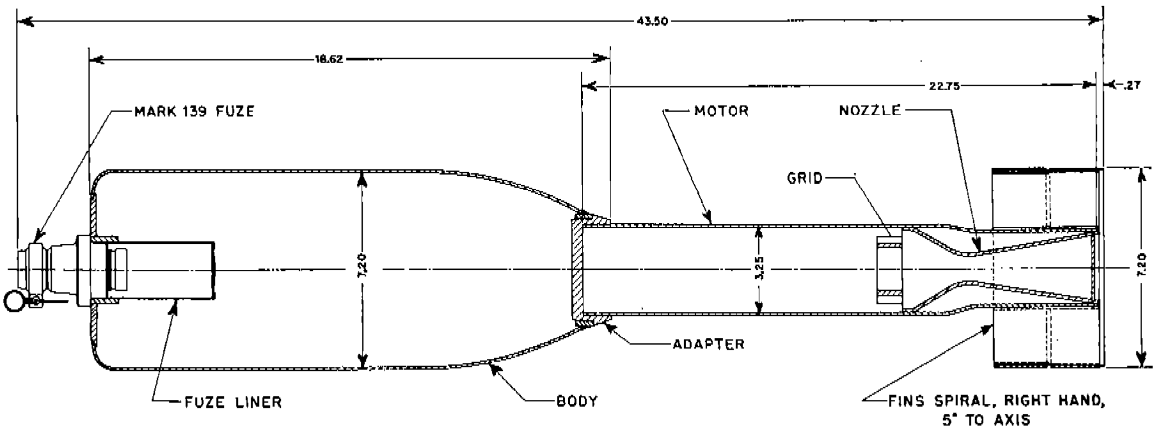

FIGURE 2. Section drawing of one of the VAR series. Others differ only in length of motor tube and nozzle diameter.

Grains. To minimize the forward travel of the rockets, a short burning time was desired, and after static tests a web thickness approximately the same as that of the ASR grain was settled upon.

19.1.3 Launchers and Service Use

The first and largest launcher installation developed for VAR's was that on the PBY-5A or Catalina flying boat. It consisted of 24 channels, 12 under each wing, formed from $\frac{1}{8}$-in. Dural sheet, the individual rails being fanned outward by varying amounts to spread the impact pattern. The rockets were hung under the channels on lug bands which rode on the turned-in edges of the channels. An intervalometer was inserted into the firing circuit so that the rockets were fired in three symmetrical groups of eight to give an impact pattern approximately 140 ft wide and 40 ft along the direction of flight.

The next plane equipped was the B–18, which

[redacted]

carried 16 steel launchers similar to those on the PBY. The original launcher design for this airplane was a Dural tube in which the rockets fitted fairly snugly without requiring lug bands. These were objectionable because they caused much more drag on the airplane than the flatter channels but principally because the aspirator effect of the nozzle reduced the air pressure behind the rocket so much that the burnt velocity was reduced as much as 20 per cent in some cases. The B–24 was also equipped with launchers on the fuselage. Neither of the Army planes took the VAR into combat, however, because sole responsibility for aerial antisubmarine warfare was soon assigned to the Navy.

The PBY and the TBF (carrying 4 launchers on the fuselage) actually used the rockets against enemy submarines with good effect. Vertical bombing proved to be much less useful than had been expected, however, because of a change in German submarine tactics. Vertical bombing theory assumed that the submarine would be submerged or getting there as rapidly as possible, but, during the latter part of 1943, German subs began staying on the surface and fighting it out with their deck guns. In this situation it was too dangerous for the attacking plane to make a straight run at low altitude as was required for vertical bombing. Only in special areas, such as the Straits of Gibraltar, where submerged submarines attempted to slip into or out of the Mediterranean, were the potentialities of the VAR–MAD combination fully realized.

As development of various installations proceeded, the emphasis shifted gradually from vertical to retro bombing. Firing backwards with a velocity considerably exceeding that of the plane had the advantages that (1) the rocket was more stable so that yaws on striking the water were smaller, (2) the launchers could be pointed slightly downward or even at a considerable angle so that flight time and hence dispersion could be reduced, and (3) firing could be delayed until the plane was somewhat past the target, thus simplifying the sighting problem. Considerable experimental work was done with the BR from the A–20 and the B–18, but no service requirements for the installations materialized. Photographs and brief discussions of the installations are contained in the summary reports on retro bombing,[1,3] but further details can be found only in the weekly progress reports of the period.

Tests of retro firing of 100-lb bombs propelled by six ASR motors were also carried out.[4]

19.1.4 Reports

References 3, 5, 6, and 7 give a complete account of the progress of the vertical-bombing program from beginning to end, discussing both the ammunition and the installations. A bibliography of various other reports pertaining to particular installations is given in reference 3.

19.1.5 Related Rockets

Although the VAR motors did not find extensive use in the application for which they were originally designed, they were adapted by Army and Navy Ordnance for various other purposes. An example of this is the "Cutteroo Grapnel," to propel several multipronged hooks and a steel cable. When the hooks were pulled back, they could clear out barbed wire, detonate land mines, or do other jobs. A similar use of the motors, in which CIT was directly involved, was in obtaining samples of the earth from the center of the crater at the first atomic bomb test in New Mexico, July 16, 1945. The existence of three motors differing in thrust but being otherwise interchangeable made the development of such uses relatively simple.

The first model of the 3.25-in. Aircraft Rocket Motor, the Mk 6, utilized the same nozzle design as the VAR's but with a different tail. It is discussed more fully in Section 19.2.2.

The first "window" rocket (3.25-in. Rocket Mk 4 Mod 1) used the Mk 2 VAR motor intact except for the tail, which was cut down from the tail of the Mk 6 motor to enable firing from T-slot launchers. The only serious design problems in connection with the window rocket had to do with the ejection charge in the head, which has been discussed in Chapter 16. The purpose of the rocket was to eject at the peak of its trajectory its load of metalized paper strips to confuse enemy radar. Although the range of the rocket was rather short and the reliability of the ejection charge not all that could be desired, it was effectively used at the time of the Normandy landings and later.

As discussed in Section 18.4, the final model of the *chemical warfare rocket* [CWR] used the Mk 3 VAR motor with a thicker-web charge.

The Mk 3 motor with a special tail was used also for the smoke float rocket, which was developed in 1943.[8,9]

19.1.6 Drift Signal Rockets

As soon as it had been shown that vertical bombing of submarines was feasible and the tactics began to be worked out, it was apparent that a means of marking the submarine's position was required. The marker should be rocket-propelled so as to duplicate the trajectory of the vertical bombs themselves. The patrol plane could then cruise at low altitude over the ocean, and, if a magnetic signal like that of a submarine was received by the MAD, it would fire a marker. By making several passes over the sub and releasing a marker at each contact, its course and speed could be estimated so that the actual bombing attack could be made with the best chance of success.

Standard Navy drift signals were to be used, the weight of which was only about 4 lb (the one finally adopted weighed only 3 lb), so that sufficient velocity could easily be attained with 1.25-in. motors. The problem was to get the right velocity to match the trajectory of the VAR and to ignite the flare heads. The first models utilized the *chemical warfare grenade* [CWG] motor with the 4-spoke charge (see Section 18.6.1), and rather extensive static-firing tests of various lengths of grain were made, since velocities as low as 35 knots were under discussion. It was found that the shorter grains required higher nozzle K's for satisfactory burning, even as high as 250, and that radial holes greatly improved the burning characteristics of the multi-web grains just as it had for the tubular. Percussion ignition was used on the earliest models, a spring-loaded firing pin firing a .32-caliber blank cartridge into 4 g of Quickmatch in the front end of the motor. This was soon abandoned for electrical ignition, however.

The short burning time given by the 4-spoke grain was no advantage for this motor, and it was more complicated to extrude. A 1.0 x 0.5-in. three-ridge tubular grain was therefore specified, and it became standard for all 1.25-in. motors. Trial of several nozzle designs resulted in the adoption of the swaged-in machined nozzle shown in Figure 3B of Chapter 23, and it, too, became standard. Ignition of the flare head was easily accomplished by letting the thrust of the motor shear a wire, allowing the motor to move forward in a sleeve and strike a blank cartridge in the rear of the head. After the thrust ceased, motor and head were separated by a spring, since the weight of the motor would otherwise sink the head. The burning time of the motor was so short and the ammunition dispersion so small a fraction of the overall dispersion that no tail was found to be required on the motor.

During the course of bombing experiments at the Goldstone Range in the Mojave Desert, one important lesson was learned the hard way when a motor ejected its nozzle, which pierced the closed breech of the launcher, narrowly missed the man who was loading it, went out through the wall of the cabin, and entered the gasoline tank in the wing of the PBY, where it was brought to rest by the high drag of the gasoline. After the several hundred gallons of precious fuel had drained out and evaporated, the nozzle was recovered and found to be neatly plugged by a little tube of cellulose acetate, the squib compartment of the molded plastic igniter which had just become standard a few days before. Thereafter igniters for small-nozzle rockets were carefully designed to avoid any possibility of large fragments.

FIGURE 3. Drift signal rocket in Mk 2 launcher.

In CIT reports, these rockets were first called *vertical flare bombs* [VFB] or *vertical flare rockets* [VFR] and later *vertical float lights*. Earlier Navy designation was Drift Signal Rockets Mk 15 and Mk 16, but the latest nomenclature drops the Mark numbers and specifies them by velocity—200 fps and 300 fps. Other slower models were worked on but not standardized.

Launchers

The tactics of submarine hunting with aircraft required that it be possible to fire a considerable

[redacted]

number of drift signals, and a reloadable launcher was therefore indicated. A closed-breech launcher was designed with a loading door on the side at the rear. It projected backward and about 15 degrees downward through a hatch in the under side of the plane near the tail. The flare head ran on guide rails inside the main tube with about 1 in. clearance on all sides so that no pressure built up inside the launcher. After the accident at Goldstone, the breech of the launcher was reinforced with a steel plate. A twin launcher of this type (see Figure 3) was adopted as standard and designated the Aircraft Rocket Launcher Mk 2 Mod 0.

The drift signal rockets were made exclusively for vertical bombing, and their service record is therefore identical with that of the VAR's.

REPORTS

The drift signal rocket is discussed in most of the reports on 7.2-in. retro rockets (see Section 19.1.4). See also reference 10.

19.2 3.5-IN. AND 5.0-IN. AIRCRAFT ROCKETS [AR]

19.2.1 Development History

When the CIT group began, the development of medium- and high-altitude antiaircraft rockets was one of the principal projects assigned to it, because the development of such a high-performance rocket would necessarily depend on the development of techniques for dry-extruding very large propellant grains. A few field tests of such rockets were made in the early days of the project, but little progress could be made until the 8-in. extrusion press was put into operation in April 1942, and by this time the ASR, BR, and other rockets had taken a higher priority. Some work on high-performance motors went on during 1942, but it was mainly with 2.25-in. motors because grains and metal parts could be produced cheaply in this size.

In the early spring of 1943, with the virtual completion of experimental work on the BR, the problem of designing a 3.25-in. motor with as large as possible a propellant grain was attacked with vigor. This caliber was chosen because dies for extruding tubular grains were available and because it was desired to duplicate with ballistite the performance of the British cordite-propelled UP3, which had been put into service in 1941.

During March and April, static and field tests were made with CIT-extruded tubular grains of cordite and ballistite in 14- and 11-gauge motors and the following results were established:

1. With either propellant, a refractory coating is necessary on the interior of the 14-gauge tubes, since otherwise heat failures are experienced at high temperatures.

2. The thickness of the 11-gauge tubes is sufficient to make refractory unnecessary with up to 6.8 lb of ballistite, but a fairly small increase in burning time might make the motor unsafe because of heating.

3. In static firing, grains of 2.5 x 0.4-in. three-ridge ballistite weighing 6.2 lb were satisfactory up to 130 F with either rod stabilization or radial holes, and 6.8 lb was satisfactory with rod stabilization. In the field with the addition of the setback force, the 6.2-lb rod-stabilized grain was satisfactory at high temperatures, but the other two were not.

In May the Bureau of Ordnance requested development of a ship-to-shore rocket to have a range not less than 10,000 yd with any of three interchangeable heads: (1) a light-case head for chemical, smoke, or high explosive for blast effect, (2) a high-explosive fragmentation head, and (3) an incendiary head. The motor was specified as 3.25-in. diameter, and, although the head weight was not specified in the directive, initial experimentation was conducted with a head having the weight of a 75-mm shell, approximately 13 lb. It was found that the rod-stabilized 6.8-lb grain would achieve the required range but the 6.2-lb grain would not. Neither was satisfactory, however, because the stabilizing rod was eroded through and ejected white-hot near the end of burning.

Meanwhile, when the difficulties involved in increasing the weight of a tubular charge had begun to become apparent, the propellants section had commenced work on extrusion dies for a cruciform charge,[11,12] the British having had good success with a grain of similar shape. After the technique of inhibiting these grains and the proper arrangement of inhibiting strips had been worked out, a 9-lb cruciform grain gave excellent performance statically even at 140 F. This weight was sufficient to give 10,000-yd range to a 20-lb head provided that the fuze had a small enough drag.

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Early field tests of the upper temperature limit of this grain were complicated by the fact that the grid seating surface at the front end of the nozzle was too narrow, so that, when motors burst and the recovered grids and nozzles gave evidence that the grid had slipped, it was impossible to determine whether the grid slippage had been the cause or the result of the high pressure which burst the tube. Numerous failures at both 120 F and 130 F were experienced, and even yet the full explanation for them is not known. They nearly always happened after the motor had left the launcher, but the time of burst varied all the way from less than one-third burnt up to nearly seven-eighths burnt. Failures occurred at the nozzle end, in contrast to the behavior of all other motors, but was not the result of heating because the camera records showed that they were preceded by a definite increase in acceleration and in the luminosity of the jet, presumably because broken pieces of powder began to be ejected and to burn outside the nozzle. One piece of grain was recovered with a piece of inhibitor strip attached which showed that the front ends of the inhibitor strips were completely eroded away before the middle of the burning time, and this was immediately confirmed by partial burnings with 11-gauge motor tubes, previous firings having failed to disclose it because they were made with thicker tubing which did not heat up so much. This experience taught us another important lesson, that static-firing tests should be done with completely standard motors.

Even after the inhibitor strips had been increased from 0.05 to 0.10 in. in thickness to prevent their eroding away and decreased from 8.5 to 7.5 in. in length so as to make the burning more regressive and reduce the end pressure peak, and the grid seating surfaces had been made adequate, occasional bursts at 120 F, frequent bursts at 125 F, and about 80 per cent bursts at 130 F occurred. An increase in nozzle diameter from 1.44 to 1.50 improved the performance greatly, giving only one burst out of 33 rounds at 130 F and none out of 100 at 120 F. Nevertheless, in proof firing a few weeks later, one motor burst at 120 F. This burst impelled the decision, which had previously been contemplated, to reduce the grain weight from 9 to 8.5 lb in order to raise the upper temperature limit to the point where the rockets could be proof-fired at 130 F and approved for service use up to 120 F. A somewhat later change in motor design, which reduced the internal K slightly, gave still better high-temperature performance, as mentioned later.

The adoption in September 1943, of the 8.5-lb grain, later designated the Mk 13 grain, solved the most difficult problem of the 3.25-in. AR motor. While the propellants and motor design groups had been preoccupied with this problem, other developments had been taking place which had changed the nature of the rocket drastically. The British success in adapting the UP3 to antisubmarine use from aircraft strongly indicated the desirability of a parallel development in this country. Thus in early June, a 20-lb solid steel head had been designed, and aircraft forward-firing launcher development had begun. Forward-firing tests from airplanes in flight, first with British rockets and soon with CIT rockets, became more and more frequent. By the middle of August, the AR had been assigned the highest priority among all the antisubmarine weapons, and CIT had been requested to manufacture 10,000 rounds per month until Navy contractors could get tooled up to begin. One month later, the request had been increased to 100,000 rockets in six months, and this number was actually delivered by the end of the following March.

Although the original purpose of the 3.5-in. AR, that of puncturing holes in submerged submarines, required only a solid steel head, other uses of the rocket developed much more rapidly than the rocket supply, and other heads were designed. In particular, the base of the 5.0-in. AA common shell was boat-tailed and bored out to take a motor adapter and became the 5.0-in. Rocket Head Mk 1. The combination of this head with the 3.25-in. motor became known as the 5.0-in. AR and ultimately overshadowed the 3.5-in. AR in importance as the submarine menace declined.

19.2.2 Motor Design Features

The first motor which was extensively tested was the 3A9 (designated by its drawing number series). Since it was designed for long-range firing, it was made as smooth as possible on the exterior. At the rear, the motor tube was swaged to a smaller diameter for a distance of 6 in. to allow the tail, consisting of four radial fins attached to a cylinder (similar to the CWG; see Figure 7 of Chapter 18), to slip over it without increasing the outside diameter. The head was the same diameter as the motor tube

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and was attached by screwing into an internally threaded ring held in the motor by a piston ring and sealed with an obturator cup. The only protuberances beyond the 3.25-in. diameter were the four fins and four little buttons, two at the front and two at the rear, which, in addition to supporting the round in a T-slot launcher, held the threaded ring and the fins in position. The primary difficulties with this design were that the rather complicated front closure increased the motor loading time and the lug buttons, which were simply threaded into place, were not thought to be safe enough for aircraft use where constant vibration for long periods might loosen one of them.

When it became apparent that the principal use of the rocket would be from aircraft at relatively short ranges, where drag was no longer important but greater stability was desirable, it was decided to make the head 3.5 in. in diameter so that it could be threaded onto the motor and to increase the fin size from 3 x 6 to 5 x 8 in. The fin change necessitated a redesign of the rear end of the motor, and to use available tooling it was made identical with the VAR motors which were in production. It was thought that it might be desired to fire this motor with a VAR head and tail, but it turned out that this combination had insufficient stability and was extremely wild. The new motor was the 3A12 and it became the first standard service AR motor, designated Mk 6. With a solid steel head (3.5-in. Mk 1), it formed the 3.5-in. AR Model 1 (see Chapter 17, Figure 2).

The 3A12 was soon abandoned because the manufacturers which the Bureau of Ordnance chose to produce the rocket in quantity objected to the complicated shape of the nozzle end of the motor, and the 3A16 or Mk 7 motor was designed in close collaboration with them to make it as adaptable to quantity production as possible. The motor design involving the bead at the front end of the nozzle is discussed in Chapter 23 and is shown in Figure 4C of Chapter 23. It had an important advantage from the standpoint of ballistics also in that eliminating the swaged portion ahead of the grid decreased the internal K slightly.[a] Since the primary difficulty with the upper temperature performance of the motor was its high internal K, this was expected to alleviate the difficulty. An interesting analysis of the actual effect of the change is given in a weekly progress report.[13] The data must be qualified with the statement that the number of rounds fired at the extremely high temperatures was not sufficient to give a low probable error, but, taken at their face

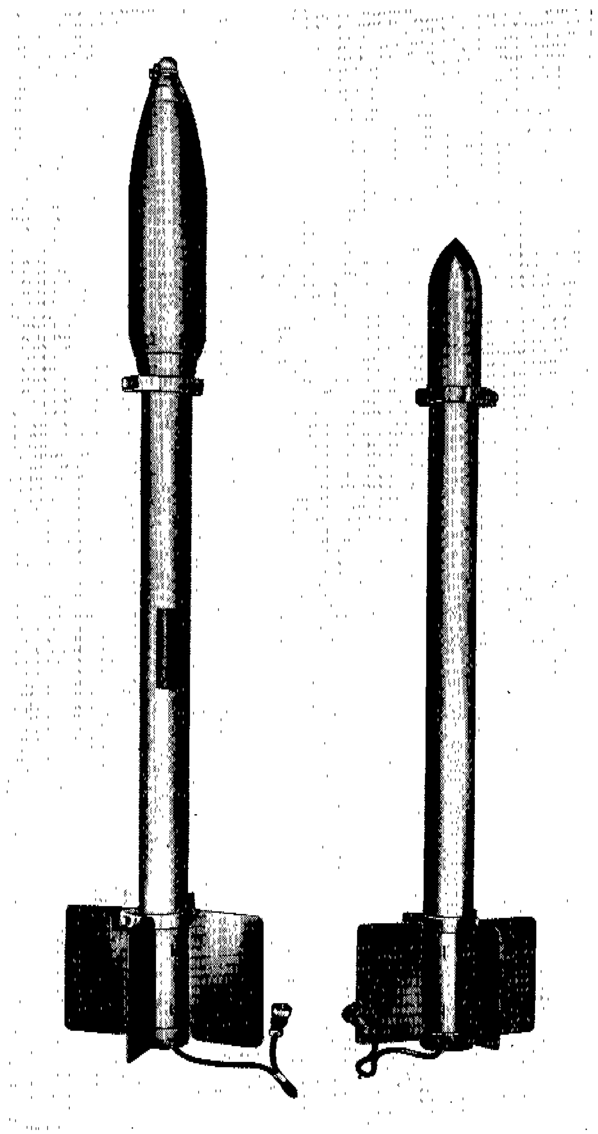

FIGURE 4. 5.0-in. AR Model 12 as fired from T-slot launchers and 3.5-in. AR Model 5 as fired from post launchers.

value, they show that the probability of a motor burst with either motor increases very rapidly above 140 F but is less than half as great for the 3A16 as for the 3A12.

The Mk 7 design was found to be quite satisfactory, and production of the motor ran into the

[a] Internal K is the ratio of the burning area of the grain to the port area around the grain through which the gas must escape. See Section 22.4.2.

[redacted]

millions, CIT contributing the first one-tenth million. The first part of CIT's production necessarily used welded tubing, and, despite pressure tests on the motor, occasional bursts during firing occurred along the weld line. For its own and the latter part of CIT's production, the Navy procured seamless tubing of NE 8735 steel, which had more than adequate strength and eliminated this difficulty. This considerable increase in tubing strength left the nozzle the weakest portion of the motor, with the result that occasional nozzle failures were experienced in Navy proof firing at high temperature. There were three possible causes of the failures: (1) The nozzle exit cone may have had a very thin spot on one side so that the gas pressure in the annular space between it and the nozzle bulged it inward. (2) The end of the nozzle skirt may not have been brazed securely to the tube in one section so that again the pressure could bulge it inward and tear it away from the tube. (3) The grain may have been faulty so that the motor pressure simply rose to such a value that the weakest point had to yield even though it may not have been faulty. To the writer's knowledge, it was never determined which was the cause, although it was the opinion of the projectile section at CIT that (3) was the most likely.

A multinozzle design for the AR motor was extensively tested and laid the groundwork for the design of the HVAR motor. Six nozzles in a circle and a central blowout nozzle were machined in a steel nozzle plate which was threaded into the motor tube. Carefully made motors of this type gave a dispersion of approximately 15 mils, which is a considerable improvement on the standard model. In reply to a request from the Bureau of Ordnance for a nozzle design that would eliminate the failures occurring in proof firing, the multinozzle design was recommended by CIT, but it was never put into production.

Grids. The obvious shape for a grid for a cruciform grain was cruciform, and once the thickness of the four arms (½ in. wide and ¾ in. thick) had been determined by a few static tests, no further change in grid design was made except to reword the specifications slightly whenever anyone thought up a new and better method for manufacturing them. The only difficult grid problem was how to hold the grain on it so that it would not rotate. The earliest method was to rivet a celluloid end washer to the grid and cement the grain to the washer with cellusolve. An immediate improvement on this design was to cement a second washer to the grain and then cement the two washers together, thus protecting the rivet heads from any possibility of erosion. The indexing pins (see Figures 15 and 16 of Chapter 22) were soon adopted as simpler and perhaps surer, although there was no evidence of failure of the other system.

The so-called "button grid," a design which would make it unnecessary to orient the grain in a particular way, was extensively tested. The grain rested on a steel disk, 1.38 in. in diameter, supported on the nozzle by a spider. The legs of the spider were far enough removed from the end of the powder grain so that adequate clearance for the gas passage was provided even though the spider legs might fall directly between the arms of the grain. Static and field tests showed only negligible differences in performance between this and the standard grid, but partial burnings showed that, because of the smaller bearing area of the button than of the standard grid, the end of the grain was deformed around the button. Since the difficulty at the upper temperature limit was almost certainly due to too great forces on the grain, it was believed that the decreased support would surely reduce the upper temperature limit. Not enough rounds were fired to confirm or refute this belief, but it did appear that the effect was relatively small. For shorter cruciform grains where the forces are not so great, button grids appeared very promising and were later used in the spinners.

Lug Bands. The 3A9 motor had the threaded button lugs as already mentioned, and the original lug bands for the 3A12 were of Dural with a riveted button of the same shape. These simple lugs were satisfactory because the T-slot launcher was shaped so as to bear on the cylindrical portion of the band or motor to provide the sway bracing. The fabrication of the launcher in this shape was difficult, however, so it was decided to make the launcher surface smooth and put the sway braces on the lug bands. Large numbers of the 3A12 lug bands (so-called "3.0-in. lug bands") were on hand, and they were adapted simply by riveting little steel "ears" on the Dural band. With the appearance of the 5.0-in. head, the 5.0-in. lug band was designed and became standard for all motors. The strange shape of the clamping mechanism on the 5.0-in. band was intended to balance the large air drag of the opposite side of the band. It did not prove to be a very good design, but the one illustrated in Figure 12 of Chapter 23 was not thought of until a year later. Since the bands were designed to fit the Mk 4 launcher which became almost immediately obsolete, they were not very well adapted for the zero-length launcher and should not be considered a model to be copied. To fit the rear post of the zero-length launchers, the "tunnel" lug band was designed. Its large "ears" serve the purpose of holding the tail fins in proper orientation, as does the sway-bracing structure of the 5.0-in. band (see Figure 5).

FIGURE 5. Front and rear lug bands for 5.0-in. AR with 3.25-in. motor as fired from post launchers.

Tails. The simplest design of a radial-fin tail is to form each fin and one quarter of the cylinder in one piece and weld the four identical pieces together, and this was the method adopted for all 3.25-in. AR motors. Since the motors were light enough to be packed four in a box with four fins nested between them, the bulk of the tail assembly did not appreciably increase the shipping volume. The 3A12 tail had a threaded ring which screwed onto the rear of the motor tube and was held with a set screw. The 3A16 design was much more satisfactory, being simply slipped onto the tube and held by the tail ring which screwed on separately, and the 5.0-in. lug band fitting between two adjacent fins kept the tail from rotating out of position. The primary difficulty with the tail was that the bumped-in portions (between the slots which can be seen in Figure 4) were not always made the proper depth in quantity production so that many tails were excessively tight and difficult to assemble. It was aggravated by the fact that the 8735 tubing tended to have a larger diameter than the original tail dimensions had contemplated. In addition, the kind of handling to which rockets are subject in service resulted in the fins being rather frequently bent, causing wild dispersion. The double-fin design of the HVAR (see Section 19.4.1) was much preferred in this regard.

Electrical Contacts. It was originally expected that large numbers of British RP–3 would be used in antisubmarine warfare, and it was therefore desirable to have the two rockets as nearly interchangeable as possible. The British were using a large and rather complicated electric plug for attaching the igniter leads to the launcher, and a simpler die-cast version of it was adopted as standard for American aircraft rockets. The plug was not very satisfactory; it was bulky, not waterproof, and easily damaged, and in addition it took too long to attach it to the launcher. Near the end of World War II, plans were made to replace it with a smaller plug which would avoid the difficulties and which would become standard for both Army and Navy rockets, but the change had not been accomplished when CIT ceased production.

Caps. Because of the weight of the grain, it was deemed desirable to provide more positive support for the front end of the grain than was given by the fiberboard seal. A die-cast cap was designed, which threaded onto the front end of the motor, and in the space between it and the seal were inserted a length of cardboard tubing and enough perforated cardboard washers so that the cap would absorb the impact of the grain if the motor were dropped on its nose. The thermal expansion and contraction of the grain was provided for by a thick felt washer inserted between the seal and the igniter. As an additional safeguard against moisture, it was desirable that the front cap be fairly watertight, but yet it

[redacted]

should not hold more than about 50 psi internal pressure so that the motor would not be propulsive in case of accidental ignition. An attempt was made to groove the bottom of the cap so that it would blow out at low pressure, but this proved to be difficult and to require too close tolerances. The bottom was punched out of the cap, and it became merely a threaded ring which held a flat steel disk and a fiber washer against the end of the motor tube. This system was satisfactory.

On the nozzle end of the motor, a drawn steel cup, held in place by the tail ring, also acted as a secondary moisture seal and held the electric plug. Rendering it nonpropulsive was a more difficult problem than for the front cap, and it was finally solved by the blowout patch, which later became a standard seal component (see Figure 13 of Chapter 23).

19.2.3 Heads

The first head to be used in service was the 3.5-in. Mk 1 (Navy production Mk 2) copied from the British head for use against submarines. It is shown in Figure 4 of this chapter and in Figure 3A of Chapter 15. It was replaced by the double-ogive Mk 8 (see Figure 3C of Chapter 15) after the latter had been shown to have a much longer lethal range, and in fact the British also adopted it. Two other 3.5-in. heads deserve mention although their service use was, to the best of the author's knowledge, very limited. They are the Mk 6 smoke head (BuOrd Mk 9) and the Mk 3 high-explosive head (BuOrd Mk 5). Both carried too small a payload (9.4 lb of FS or 2.3 lb of TNT) to be very useful. Probably more AR's were fired with the 5.0-in. Mk 1 head than with all others. Earliest models had nose fuzes only, but the later practice was to supply them with a PIR base fuze and a SAP steel nose which could be replaced by a nose fuze in the field. By means of the fuze-arming solenoid, the rocket could be fired so that either the nose fuze or the base fuze functioned, depending on the type of target.

19.2.4 Fuzes

The earliest nose fuze, the Mk 148, was adapted from the Mk 137 BR fuze by using a smaller propeller, a protective cap which was removed when the rocket was loaded on the plane, and an adapter to fit the threads in the fuze liner. As soon as production could get under way, it was replaced by the Mk 149 (see Figure 4) which was specifically designed for aircraft rockets and has a streamlined body and a waterproof cap assembly which covers the propeller and protects the working parts of the fuze from weather and icing until it is fired. It has also an acceleration-actuated shutter-locking pellet which delays the completion of arming until the end of burning. The first base fuze, the Mk 146 with no delay, was later replaced by the Mk 157 with 0.02-second delay.

19.2.5 Launchers and Service Use

The forward-firing launchers have been described in Chapter 17 and their use in service was so extensive and so well publicized that there is no reason for saying much about it here. The first submarine kill in which the AR was used was in the Atlantic on January 11, 1944. In this and in most subsequent submarine attacks, however, it was difficult to assess accurately the effect of the rockets because machine guns and depth charges were also used and because, as one Navy report slyly remarked, "the survivors never survive so that they can be questioned." The first use of the 5.0-in. AR in the Pacific was in a strike against Rabaul by Marine Squadron VMTB–134 which, unexpectedly finding itself in possession of 20 sets of Mk 4 launchers, had equipped its TBF's without the aid of any instructions, located a shipment of rockets, rescued it from the ship's hold by cutting through a bulkhead rather than unload the ship, and then trained themselves for three days. Although theoretically rendered obsolete by the HVAR, the 5.0-in. AR continued to be used in large quantities in the Pacific until the end of World War II because the HVAR was not available in sufficient quantity until the spring of 1945. It was most successful against point targets: AA positions, ammunition dumps, oil storage, planes on the ground and in revetments, small buildings, and shipping. It was particularly effective against shipping, including destroyer escorts, and it is on the record that rockets even sank one full-size destroyer. In the Iwo Jima and Okinawa operations, besides the uses just outlined, rocket-firing planes were frequently called on for ground support, especially against Japanese caves.

Surface-fired 5.0-in. AR's were also used for barrage where longer ranges than that of the BR were required. The T-slot Rocket Launcher Mk 30

[redacted]

Mod 0 (essentially identical with the CIT Type 31C shipboard launcher shown in Figure 6) saw some service in this application, notably on the LSM–R. PT boats and LCI's also fired them from Mk 4 launchers rigged up in a supporting frame. For the Okinawa operation, in addition to other craft with automatic 5.0-in. spinner launchers, eight LSM's were equipped with 480 Mk 4 launchers in close array loaded with 5.0-in. AR's and were used to good effect, although one was put out of action by a suicide plane.

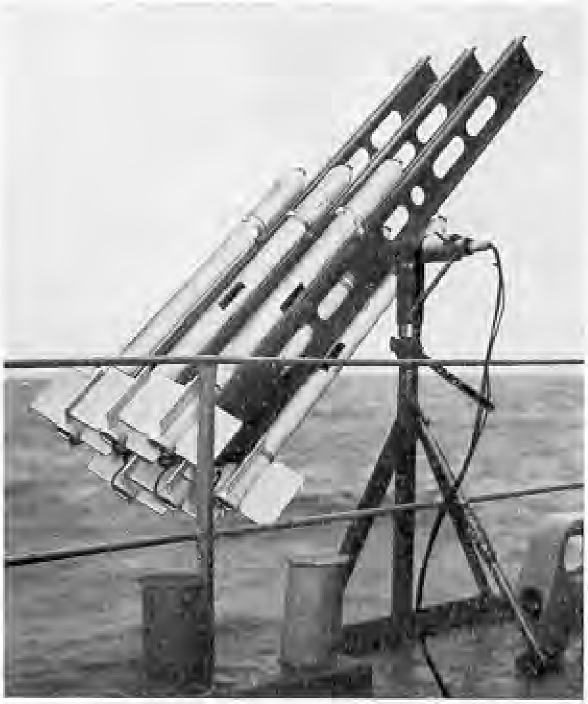

FIGURE 6. Type 31C shipboard launcher loaded with 3.5-in. AR's with Mk 3 heads.

19.2.6 Designations and Types

With two different motors, five or six different heads, and five different fuzes, if the SAP nose is included, the nomenclature required to keep all the possible combinations straight becomes rather involved and will not be given in detail. The CIT designations most often met in the literature are

3.5-in. AR Model 1—Mk 6 motor with Mk 1 head.
3.5-in. AR Model 5—Mk 7 motor with Mk 1 head.
3.5-in. AR Model 14—Mk 7 motor with Mk 8 underwater head.

Latest Navy designation is 3.5-in. Rocket Mk 1 Mod 0 for all rounds with solid heads, Mk 1 Mod 1 for all with TNT heads, and Mk 3 Mod 0 or Mod 1 for all with smoke heads. The Mk 7 motor with the 5.0-in. head is 5.0-in. Rocket Mk 1 Mod 0 with either of the nose fuzes but no base fuze, Mk 1 Mod 1 with nose fuze and Mk 146 base fuze, Mk 1 Mod 2 with nose fuze and Mk 157 base fuze, and Mk 1 Mods 3 or 4 with no nose fuze but Mk 146 or Mk 157 base fuze. A more complete list of designations is given in *Ballistic Data*.[2]

19.2.7 Reports

The complete development of the 3.5-in. AR is sketched in two CIT reports: references 14 and 15. The latter contains a complete index to the weekly progress reports and bibliography of reports issued up to the time of its publication. Reports on underwater tests and trajectories include references 16, 17, 18, and 19. Pilot production methods are discussed in references 20 and 21. The more important reports on forward-firing ballistics include references 22, 23, and 24. The use of the rockets in forward firing is discussed in references 25, 26, 27, 28, and 29. In addition, there are a large number of reports entitled *Forward Firing of (Blank) Rockets from (Blank) Aircraft*, issued by the Army, Navy, and CIT.

19.3 2.25-IN. AIRCRAFT ROCKETS [SCAR]

The development of the 2.25-in. *subcaliber aircraft rocket* [SCAR], usually pronounced "scar," began in January 1944. Since the purpose of the rocket was training pilots in firing the larger aircraft rockets, it would have been desirable to duplicate the standard trajectories exactly. This was realized to be impossible, however, since a small rocket would necessarily have a considerably larger deceleration coefficient because of its small weight, a shorter burning time, and a different variation of velocity with temperature. The specifications therefore called for subcaliber rockets to duplicate as nearly as possible the trajectories of the 3.5-in. and 5.0-in. AR's at 70 F, 20-degree dive angle, 230-knot airspeed, and 1,000-yd range, these being conditions which were frequently used in training. To simplify manufacturing, it was desired to use the ASR–BR nozzle without modification if possible.

To match the 1,120-fps velocity of the 3.5-in. AR in the 2.25-in. caliber even with no payload required a high-performance motor unless unusual measures were taken to lighten the motor tubing. Preliminary calculations indicated that 1.85 lb of propellant would be needed. Considerable experience with grains in this weight range had already been acquired. In the summer of 1942, attempts had been made to increase the length of the ASR grain above the standard 11.6 in. Tests were made on 14-, 16-, and 18-in. lengths, and even on the shortest it was found impossible to get satisfactory performance above 120 F with the 1.7 x 0.6-in. powder. Thicker-web grains, 1.7 x 0.25-in., worked better, but on a projectile like the ASR or BR their longer burning time would greatly decrease the accuracy. Throughout 1943, experiments on 2.25-in. motors, usually with thinner walls than 11 gauge, were carried on to learn about the factors which determined the amount of powder which could safely be used, and a 2.25-in. rocket, sometimes called an antiaircraft and sometimes an antitank rocket, was standardized.[30] With no payload, velocities as high as 2,600 fps had been achieved with it.

The restriction to 11-gauge tubing brought the attainable velocity of a 2.25-in. rocket down to the neighborhood of that actually required, and, when the SCAR was first proposed, there was some doubt as to whether a tubular grain could be used satisfactorily. The propellant problems were solved successfully without recourse to special grain shapes, however.

By April 1944, CIT production of metal parts for Navy use was in excess of 1,000 per day and of complete loaded rockets in excess of 300 per day. The rate of metal parts production soon doubled, and total production was more than 200,000. The Navy's own production was, of course, many times greater.

19.3.1 Types and Designations

The Model 1 SCAR, intended to duplicate the 3.5-in. AR trajectory, has an overall length of 29.2 in., of which 26 in. is motor, the head being simply a hollow streamlined motor closure. Its grain weighs 1.75 lb. The motor is Mk 10 or Mk 11 according to whether it was produced by CIT or BuOrd and Mod 0 or Mod 1 according to whether it has a screw-in or a formed brazed-in nozzle. Heads are Mk 1 or Mk 3. All variations are designated 2.25-in. Rocket Mk 1 Mod 0.

For matching the 5.0-in. AR, the simplest procedure at first appeared to be to use the same motor with a heavier head, and more than 10,000 of these rockets, the CIT Model 2, were manufactured. The opposite alternative, using the same head but a different motor, was soon adopted, however, as the Model 3. It differs from the Model 1 only in having a nozzle throat small enough to accommodate its 1.12-lb grain. It was made only in the formed-nozzle version, motors Mk 12 and Mk 13. All varieties of the slow SCAR are designated 2.25-in. Rocket Motor Mk 2 Mod 0.

19.3.2 Design Features

Grains. The first calculations indicated that 1.85 lb of 1.70 x 0.26-in. ballistite would be required to give the necessary velocity. This grain gave an internal K in excess of 150, so that, as was expected, static firing indicated that variations in external diameter gave large differences in performance. Thus at 130 F, a grain with an external diameter of 1.71 in. gave a maximum pressure drop along the grain of 285 psi, whereas a grain only 0.02 in. larger gave 450 psi. If the outside diameter were carefully controlled, it appeared from static tests that the grain would probably be satisfactory up to perhaps 100 F. Effective gas velocities of this charge in field firing were higher than expected, so that a reduction in charge weight was possible. After tests with 1.70 lb, which had too low a nozzle K (the nozzle diameter being fixed as that of the BR and ASR) and hence gave low gas velocities and poor low-temperature performance, a 1.75-lb charge was standardized as the Mk 16.

Occasional difficulties in low-temperature static proof firing of the Mk 16 grain together with the fact already mentioned that the internal K was higher than desirable for good high-temperature performance made it advisable to design a new grain which would be slightly longer and slimmer. This would give a higher nozzle K and a lower internal K, thus improving performance at both ends of the temperature scale (see Chapter 22). Dimensions were changed from 1.70 x 0.28 x 12.5 in. to 1.66 x 0.26 x 13.25 in., and the latter grain was standardized as the Mk 16 Mod 1. Although cellulose acetate end washers on the ASR and BR grains had been discarded as unnecessary, they are used on both ends of the SCAR grains, which otherwise would have been too regressive on account of their large web thickness.

The Mk 17 grain for the Model 3 is simply a shorter version of the Mk 16 Mod 1. So much difficulty with ignition was experienced with it at low temperatures, mainly because of the large empty space at the front of the motor, that the standard 12-g BR igniter was superseded by one containing 14 g of powder.

Nozzles. The BR nozzle was used first because it was a standard item already in production, but, when the Navy production began, different contractors were used so that this was no particular advantage. Consequently most SCAR's have been made with brazed-in formed nozzles because they are cheaper. The formed nozzle was patterned closely after those of the 3.25-in. AR motor, and those for CIT production were shaped cold in one piece from tubing. With these nozzles, no difference in dispersion from that of the machined nozzles could be detected. At the request of several Navy contractors, CIT ran numerous tests of nozzles made in two parts in punch presses and held together by a press fit at the throat. This method of fabrication left a small step at the exit end of the nozzle throat, just in the critical position to affect the gas flow in the exit cone, and probably for this reason SCAR's containing these nozzles had dispersions twice that of CIT rounds or more. They were thus never approved for production, although it was repeatedly pointed out that the trouble could probably be cured by a simple change in design to put the step on the entrance side of the throat. Nozzles made in two half shells (i.e., split in a plane through the axis and brazed together) performed satisfactorily in their only field test.

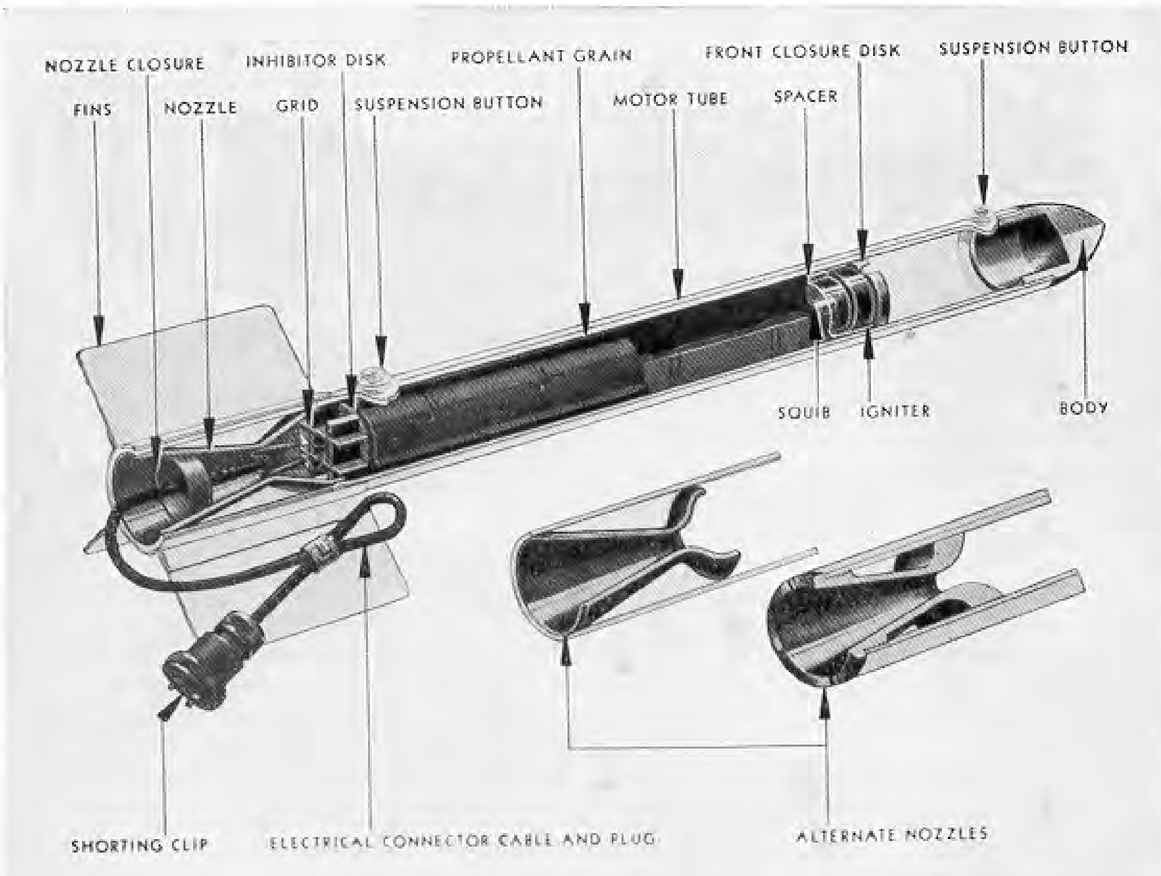

FIGURE 7. Subcaliber aircraft rocket.

Lugs. The lightness of the round made it unnecessary to provide large lugs for sway bracing, and so a specially shaped lug button was designed (see Figure 7). It had a head which rode on the top of the Mk 4 launcher slot and a wide shoulder which fitted fairly closely below the slot. It was entirely satisfactory, and the only troubles were with the method of attaching to the tube. Methods tried were (1) threading them into the tube and silver-soldering, (2) arc-welding them into unthreaded holes in the tube, and (3) attaching a special flux-filled stud with a special welding gun and upsetting the end of the stud to hold the lug button on. The last method was by far the cheapest, quickest, and generally most satisfactory, and replaced other methods for CIT production as soon as it was tried.

Fins. The radial fins are spot-welded together and the assembly spot-welded to the tube in a similar manner to the CWG. This is satisfactory since no significant saving of space would be made by having them detachable.

Heads. The original Mk 1 head was machined from steel and weighed 1.6 lb. The shortage of steel led to a request from the Bureau of Ordnance to investigate the possibility of using die-cast zinc heads. Since the head is situated where the gas stream is essentially stagnant, it was found that zinc heads do stand up satisfactorily in general, but in at least two cases a little leakage through the threads occurred and the gas eroded a hole about 1 in. in diameter before the end of burning. Although it was difficult to reproduce the phenomenon at will, it appeared that out-of-round motor tubes might cause it and that proper luting of the threads would prevent it. Accordingly it was specified that the head and motor threads be coated with a non-drying luting compound known as "Crater Compound." The zinc heads were designated Mk 1 Mod 1 and Mk 3. The heavy Mk 2 head for the Model 2 SCAR was made only from steel, and its final weight was 8.6 lb. Several other weights were tried previously in attempting to get the trajectory correct.

19.3.3 Launchers

The SCAR's were designed to be fired from the Mk 4 rails without modification, but were too short to reach between the posts of the Mk 5. Various adapter launchers were tried having different lengths from zero up to about 3 ft, since it was possible to control the tip-off and gravity drop by the launcher length and thus get the best fit to the trajectory of the standard rounds. A 2-in. constrained travel of the front lug was finally adopted, and this adapter was standardized as the Mk 6.

19.3.4 Reports

Many of the reports on sight settings, trajectories, and use of other aircraft rockets contain information on the SCAR's. The only formal reports on the rockets themselves are references 31 and 32.

19.4 5.0-IN. HIGH-VELOCITY AIRCRAFT ROCKETS [HVAR]

The 5.0-in. AR with the 3.25-in. motor was from the time of its inception admittedly a stopgap. Its velocity of only 700 fps gave it too little penetrating power and too much gravity drop and required that, to be effective, it be fired at relatively short range where antiaircraft fire was uncomfortable. In addition, its lack of stability under water restricted its usefulness as a Navy weapon. To accelerate the same 50-lb 5.0-in. head of the 5.0-in. AR to a velocity equal to or greater than that of the 3.5-in. AR required obviously a motor of larger caliber. By the late summer of 1943, extrusion presses were available which could make considerably larger grains than 3-in. diameter, and shortly after the design of the Mk 13 cruciform grain had been stabilized the propellant section began experiments on possible grains for a 5.0-in. motor, inside diameter 4.625 in.

As expected, the 5-in. grains gave the same answer as the 3-in. grains; namely, that for high loading density the cruciform shape is considerably superior to the tubular and that a spiral inhibiting pattern gives satisfactory burning curves. A 24-lb grain was designed, having a web thickness of 1.6 in. and an outside diameter of 4.22 in., so that with 0.15-in. thick inhibitor strips it was a reasonably snug fit in a 4.625-in. tube. This grain, designated Mk 18, gave beautiful neutral-burning pressure-time curves at all temperatures from −25 F to 160 F. Performance was so good that it was believed that a 20-per cent heavier grain would still work satisfactorily, but the difficulties with the 3.25-in. AR motor had taught us that it did not pay to try to push to the limit of grain size, and 24 lb was adopted as standard. This amount of propellant was more than enough to give a faster round than the 3.5-in. AR.

The first 5.0-in. motor came off the drawing board in early December 1943, and probably underwent fewer significant design changes in the course of its development than any other rocket motor. The conservative design of the charge paid ample dividends. In field firing at 140 F, even in some cases with heads weighing only 20 lb which gave considerably larger accelerations than normal, malfunctioning was so rare even with the ordinary JPN powder that this temperature was adopted for regular proof firing. Its low-temperature performance was amazing. Because of the blowout disk which enabled it to run at a K of 216 (for the 3.25-in. AR motor, $K = 167$), it practically has no low-temperature limit. To the author's knowledge, it has never been known to chuff, and even the two rounds which were packed in "dry ice" at -110 F over night showed no evidence that they were near the failure point.

From the beginning it was nicknamed "Holy Moses," obviously because at the time it appeared it seemed like such an enormous rocket. Since a number of apocryphal versions of the circumstances under which it got its name are current, the author may be pardoned for setting the record straight. It is said, for example, that "Holy Moses" was the exclamation of the first pilot who fired one. The fact is that, before it was even off the drawing board, the author gave it that name as an experiment to see if he could make it stick and become the universal unofficial name. It did.

The design and development of the Holy Moses motor (the 5.0-in. Mk 1) was completed about June 1, 1944, and CIT production was in the process of changing over from the older CIT Model 1 motor when "Project Moses" appeared on the scene. The V–1 "buzz-bombs" had just begun falling on England, and the fundamental strategy of resisting them was to eliminate the launching sites by aircraft attack, especially those in the Pas de Calais area. It was thought that the HVAR might prove an effective weapon against them, and it was suddenly decided to begin approximately five days later shipping the entire CIT production (100 rounds per day) by air to England. Fifty sets of launchers were also included, and a special mission accompanied them to England to equip a squadron of P–47 fighter planes for service-testing of the equipment. Nineteen shipments of 100 rounds each were made by air, together with one boat shipment of 500 rounds of the obsolete experimental ammunition which could be scraped together. By the time the 513th Fighter Squadron (SE), 406th Fighter Group, Ninth Air Force, AAF, was equipped and ready for training, it had been determined that the launching sites were not suitable rocket targets, and the ammunition was available for supporting the invasion of France, which it did with excellent results. In a letter of commendation written to NDRC, Major General B. E. Meyers stated that this initial use of the HVAR proved "without question the effectiveness and efficiency of this equipment in actual combat, and has resulted in providing the Army Air Forces with the best antitank weapon of the war."

The combat experience in Normandy emphasized two facts that were already known: (1) that the post launchers designed for the smaller AR's were not rugged enough for Holy Moses, and (2) that the inferior armor-piercing qualities of the head was a serious disadvantage. The AAF was sufficiently impressed, however, to adopt the rocket as standard fighter plane equipment and to undertake, in cooperation with CIT, a high-priority program of launcher development, so that by the spring of 1945 some Army fighter planes began coming off the production lines equipped to fire the Navy's HVAR.

Two views of the assembled rocket are shown in Figure 5 of Chapter 17.

19.4.1 Design Features

Tubing. NE 8735 steel was specified for the motor tubing, and, as in the case of the 3.25-in. AR motor, it was specified by internal diameter (4.625 $\pm$ 0.015 in.) and minimum wall thickness (0.187 in.). Since the tubing received averages thicker than 0.200, the motor is somewhat heavier than necessary, but this is a minor disadvantage. In other respects the tubing is almost ideal. To the author's knowledge, no motors were rejected for failing the pressure test at 5,000 psi although CIT production exceeded 100,000, and no motor bursts occurred in field firing which appeared to be the fault of the tubing. Standard motor tubes burst at pressures between 6,000 and 7,000 psi. Field tests with tubing more than 30 per cent stronger showed that increasing the tubing strength had no effect on the frequency of bursts at 160 F. Except for facing the ends and threading, the only machining on the tube is the counterbore at the front end to provide a close fit to the guiding land on the head and to the front motor seal. The first model had a longer tube and a correspondingly longer counterbore to accommodate a different head, as explained later. Because of the known disadvantages of internal motor threads, particularly V threads, pressure tests with square threads were made, but it was decided that their use would unduly complicate production and gauging.

Nozzle. The design of the nozzle and associated parts can be seen in Figure 8, and a rear view of the nozzle is shown in Figure 5 of Chapter 17. It is based upon the experience with the 3.25-in. multinozzle motor but includes one entirely new feature in supporting the grid on a grid stool in the center of the motor, thus allowing the use of a much thinner grid. The eight peripheral nozzles and one central blowout nozzle are machined in the solid nozzle plate because the nozzle area is too large to permit insert nozzles. The tooling necessary to make such a nozzle plate with sufficient accuracy and to check it for alignment is rather complicated, and during the first three months the accuracy of the rocket steadily increased as nozzle production technique improved.

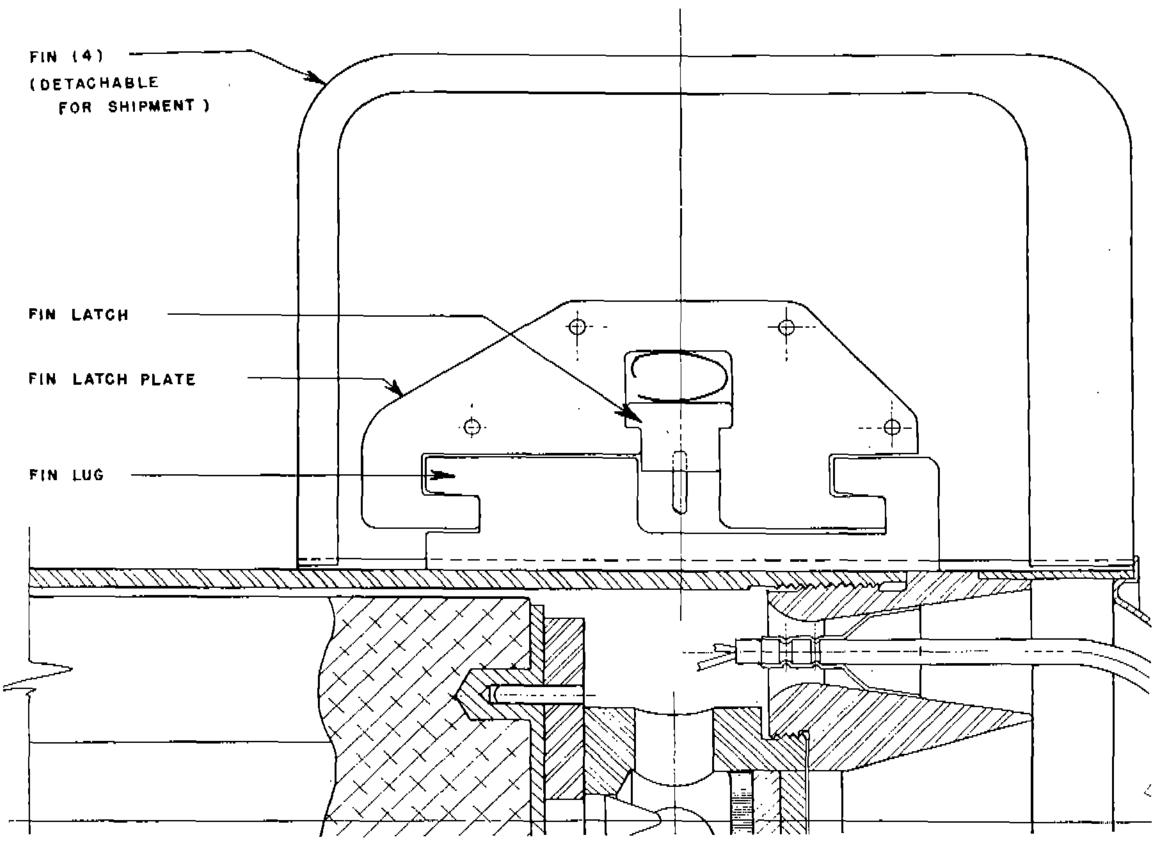

FIGURE 8. Nozzle and fin construction in 5.0-in. Motor Mk 1.

The nozzle ring or skirt extending beyond the rear face of the nozzle plate serves as a receptacle for the electric plug during shipment, but its primary purpose is to reduce the luminosity of the gas in the same way as a large expansion ratio does for single nozzle motors.

The grid stool serves three purposes: (1) it supports the grid, which is cemented to the propellant grain, (2) it clamps the blowout disk in place, and (3) it allows the motor pressure to get to the blowout disk while holding the insulation to protect it from the heat and erosion of the gas. It was found that the effective gas velocity decreased considerably when the blowout disk functioned, and it was thought that a redesign of the grid stool to allow more direct access of the gas to the central nozzle might improve it. A cast steel stool, square in cross section, was designed which has four gas access holes inclined at an angle instead of being perpendicular to the rocket axis. No difference in gas velocity could be detected, and it is probable that the turbulence of the gas flowing through the central nozzle is only part of the reason for the reduced efficiency, the lower nozzle K and nozzle coefficient also contributing. A later design than that of Figure 8 has the blowout disk in the form of a shallow copper cup which is crimped onto the grid stool, thus making it impossible to insert two disks in the same motor.

Suspension Lugs. Welded lugs were chosen in place of lug bands for a number of reasons. The spacing between launcher posts had been fixed by the 3.25-in. motor and did not appear likely to change; nor were American rockets being fired from British launchers, so that the arguments which led to the use of detachable lug bands on the 3.25-in. motor no longer held. Fixed lugs had the important advantage of rigidity and invariable spacing, and in addition they made possible an appreciable decrease in air drag. Since the Mk 4 launcher was by now obsolete, the lugs were made to fit post launchers (see Figures 3 and 4 of Chapter 17), although a small attachment was made to fit on the rear "tunnel" lug for use with the Mk 4. It was used very little, if at all, and was not even made in Navy production. Since even in ground firing, long launchers give no appreciable decrease in dispersion, there is little reason for their use.

Fins. In order to fit the same launchers, the fins were made the same size as those on the 3.25-in. motor. Detachable fins were decided upon because the motors were so heavy that they had to be boxed individually, and a one-piece tail like that of the Mk 7 motor increases the shipping volume per motor by more than 35 per cent. The fins (see Figure 4 of Chapter 17) were therefore die-formed in two pieces and seam-welded together at the edges, leaving a hollow space $\frac{3}{16}$ in. thick inside to house the latching mechanism. Details of the latching mechanism are shown in Figure 8. It was found to be quite satisfactory. The dimensions had to be worked out by trial and error, but when fins and fin lugs were properly made and not fouled with paint (a point which had to be watched), the fins were easy both to install and to remove and fitted very tightly. All the latch and lug parts could be stamped from sheet, so that they were not expensive. The four fin lugs were welded to the motor after the nozzle was installed, and no difficulties with this procedure were found.

As previously pointed out, the choice of 5 x 8 in. for the fin dimensions was arbitrary, being simply copied from the aircraft version of the RP–3. In the summer of 1945, a comprehensive test of possible HVAR fin shapes was made in the high-speed water tunnel at CIT,[33] and among the results were the following:

1. For 5-in. width, 8-in. length is very close to the optimum from the standpoint of accuracy. Ten-inch length would give very slightly more stability, but 15-in. length would be worse. In general, for any width, increasing the length beyond about 1.5 calibers gives little or no increased stability.

2. The stabilizing moment increases very rapidly with an increase in fin width. Thus 8 x 8-in. fins would reduce the yaw oscillation distance from 320 to 240 ft and reduce the dispersion from a zero-length launcher in the same ratio.

3. Tests with ring tails were made also, even though they cannot be used with post launchers. It was found that for a given size ring tails are much more efficient in providing stability than fin tails. This is illustrated in Figure 9 which shows six different tail shapes, all of which would give the same yaw oscillation distance (and hence the same dispersion, presumably) as the standard tail shown in the lower left.

4. The stabilizing moment for a given tail is quite constant for overall rocket lengths between 10 and 14 calibers, so that the results should be applicable to other shorter rockets of uniform diameter (such as Tiny Tim).

Igniters. Pending the development of a larger igniter, the earliest 5.0-in. motors contained two of the 35-g capacity plastic case igniters which were used in 3.25-in. motors. These gave satisfactory ignition but were not satisfactory for service use because they were not held securely in position. A rather heavy plastic case igniter 4.56 in. in diameter was tried. In order to accommodate two squibs and their connecting wires in a squib compartment at the rear of the igniter case, the threaded cover for the powder chamber was put at the front end.

This proved to be a fatal flaw in the design, for, when the powder ignited and bulged the case walls out until they contacted the motor tube, all the burning powder found itself confined between the front motor closure and the heavy plastic piston formed by the igniter case. The resulting force on the grain fractured it and motor tube bursts occurred at least 20 F below the temperature at which they

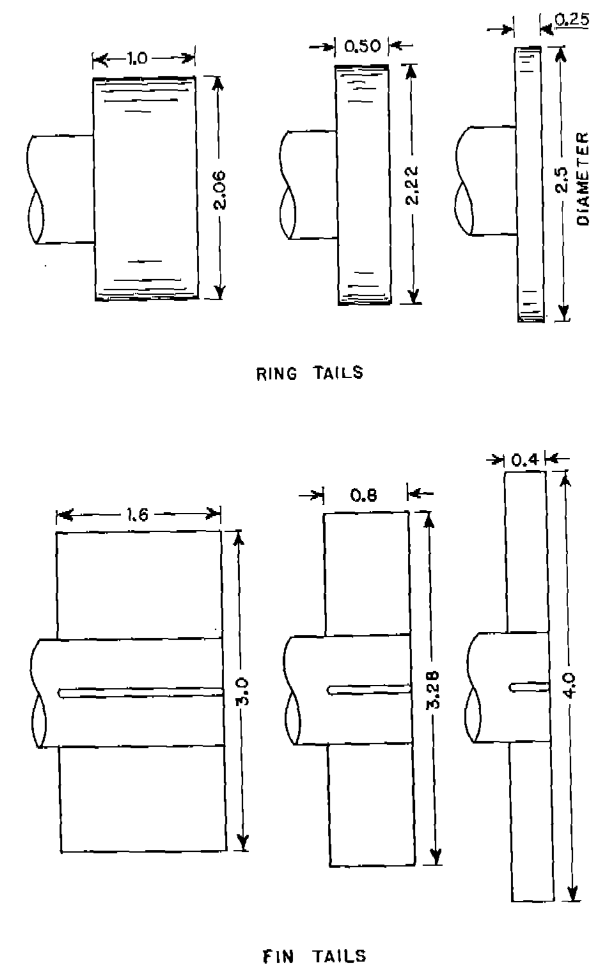

FIGURE 9. HVAR tail designs giving the same stability as the standard according to water tunnel tests.

had previously been found with the same weight of black powder in two small igniter cases.

A 55-g metal case igniter was then designed and became standard. A 70-g metal igniter was found to increase the high-temperature burst frequency slightly. The extra space which had been left for the much thicker plastic igniter was taken up by inserting a thin cylindrical steel spacer. Since length is not undesirable in fin-stabilized motors, shortening the motor to remove this waste space was neither contemplated nor tested. It was simply left as insurance against changes in propellant length. A certain amount of space is probably necessary to get good ignition, but the problem did not arise in the case of the 5.0-in. finner motor and so no tests were made of it.

When the Tiny Tim igniter was reduced from 1,200-g capacity to 230-g in order to reduce the blast effect, the question of reducing the charge in the Holy Moses igniter arose. Thirty-gram metal case igniters proved to give but little less blast, however, and were inferior to the standard 55-g igniter at low temperatures, so no change was made.

Seals and Closures. The design of a metal front end motor seal, later used in the 5.0-in. spinners and shown in Figure 13 of Chapter 23, was first developed for the HVAR. For this motor the seal had a well in the center so that the blowout patch was recessed from the front face, leaving a space for the cap or bracket on the base fuze. Glued to the seal are a 1-in. felt on the back side and a ⅛-in. felt on the front side, both perforated so as not to interfere with the blowout patch. The seal is inserted with a tool which positions it accurately so that the head, or the thread protector which extends into the motor the same distance as a head, seats against the thin felt washer and keeps the seal from shifting and breaking loose. A thin steel cup is sealed in the thread protector to provide an auxiliary seal at the front end.

The rear auxiliary seal, which as in the case of the 3.25-in. motor serves also as a receptacle for the pigtail, is pressed into the nozzle skirt ring. It was made dome-shaped in order to make it impossible to stand the motor on the nozzle end.

Heads. The first head, which eventually became the Mk 5, was made from the same 5.0-in. AA shell which had given the 5.0-in. AR its head. The only changes made on it were to bore out the base to take the PIR fuze and to thread the outside to fit the motor. For extra support in oblique water and ground impacts, the head thread was moved forward so that the base of the head extended into the motor tube and carried a "guiding land" machined to fit closely (minimum clearance 0.010 in. on the diameter) into the counterbore in the tube, which had the same diameter as the minor diameter of the motor thread. Original experimental models had a 5.5-in. overlap of the head and motor, but this was reduced to 3.0 in. because the longer overlap was believed to make the wall thickness of the head too small at one point. The final base design is shown in Figure 1 of Chapter 15 and was used on all HVAR heads.

The Mk 5 head was used because of its easy availability, and it has a number of serious drawbacks. Because of its relatively thin wall and the large hole in the nose, it does not perform well against concrete or armor plate but breaks up easily. It is also unstable under water and under ground because of too great nose lift. Three other

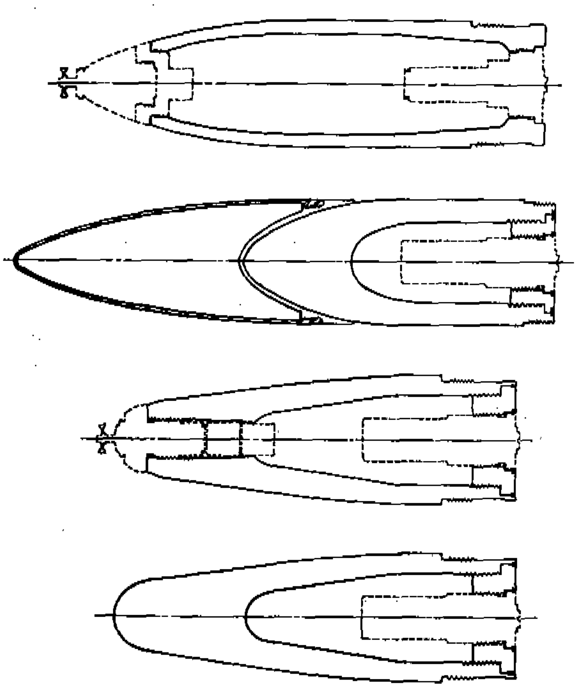

FIGURE 10. HVAR heads. *Top:* service head Mk 5 or Mk 6 with Mk 149 nose fuze and Mk 157 (PIR) base fuze (early design, prior to adoption of gas-check ring). *Below:* heads contemplated for service use. Modified Mk 46 shell (similar to Mk 2 head) with windshield and Mk 166 (DDR) base fuze. Model 31 (similar to BuOrd Type Ex-1) with AIR-12 nose fuze and Mk 166 (DDR) base fuze. Model 32 with Mk 166 (DDR) base fuze only.

heads were therefore designed and had been partially tested before the end of World War II. They are shown, with the standard Mk 5, in Figure 10. The CIT 5.0-in. Model 35, essentially the same as the BuOrd 5.0-in. Mk 2, was designed after the 5.0-in. special common projectile Mk 38 or Mk 46 for armor piercing. Preliminary tests indicated that it would penetrate 2-in. STS plate at up to 40 degrees obliquity and would probably penetrate 3-in. plate at normal incidence if the pyrotechnic delay in the base fuze were long enough. The CIT Model 31, similar to BuOrd Type Ex-1, was designed on the basis of water impact tests to have optimum underwater trajectory and was later found to have optimum underground performance as well. Although the water-discriminating fuzes originally intended for use with this head were abandoned, it would be much better for general use than the Mk 5 if it had a DDR base fuze and an instantaneous nose fuze with the same hemispherical shape as the AIR–12. In oblique impacts on fairly heavy armor plate, the nose fuze is broken off, so for some purposes the Model 31 should be replaced by the Model 32 having the same shape but with a solid nose. Although no plate tests have been made with this head, it appears likely to be very useful if made from the proper steel.

Fuzes. The Mk 149 was the only nose fuze used on the HVAR in service. Proximity nose fuzes were found to be unsatisfactory because of the prolonged afterburning, probably caused by the inhibitor strips, which are too small to be ejected through the nozzle as in the case of the 3.25-in. motor.

The nondelay base fuze Mk 146 was used first but was replaced by the Mk 157 with 0.02-second delay. When the gas check ring was adopted, the fuze became Mk 159 and a shorter delay (0.015 second) was used because of the increased velocity of the HVAR over the 5.0-in. AR for which the Mk 157 was originally designed. The Mk 159 in turn gave way to the Mk 164 which incorporates an improved shutter design to decrease the probability of duds at high impact angles. The DDR fuze, which was put into production but did not reach service use, is designated Mk 166. Description of these fuzes is given in *Rocket Fuzes*.[34]

19.4.2 Alternative Designs

Nonwelded Versions. As a result of difficulties with welding fin lugs on the Tiny Tim motor tube, a decree was laid down by someone in the Bureau of Ordnance that no welding was to be permitted either on the Tim or the Holy Moses motors. The 5.0-in. motor was therefore hastily redesigned in Washington, and the 5.0-in. Motor Mk 2 Mod 3 became the standard model for Navy production.

[redacted]

It has two lug bands similar to those on the 3.25-in. AR motor and a tail which is attached to the motor by clamping the cylindrical portion with nuts and bolts. The two-piece hollow design of the fins themselves was maintained so that the tail has adequate strength. Its chief difficulty, the amount of shipping space required, was not felt to be important.

At the request of the Bureau of Ordnance, CIT designed and made preliminary tests on another nonwelded model, which was designated only by its drawing number, 5MA4. In this design the fin lugs, rear suspension lug, and nozzle skirt are made in one assembly and attached by drive screws into the nozzle plate. This design allows the use of the individual detachable fins of the Mk 1 motor. Aside from this, the only important change was to redesign the lug band clamping system so that the band can be tightened more securely and to substitute flat-bottomed positioning holes and pins for the tapered ones which had been used on the 3.25-in. Mk 7 motor and carried over to the 5.0-in. Mk 3. These changes position the front lug band securely enough so that there is no danger of slippage under the stresses normally applied in service. No 5MA4's were produced.

CIT produced more than 100,000 Mk 1 motors without any difficulty with welding on the motor tubing. Failures occurred only at extremely high temperatures and always, as nearly as could be determined, as a result of grain failure. Occasionally such bursts showed a tendency to occur along one of the welds on the fin lugs because of the slight weakening at this point, but equally often the split ignored the welds entirely.

White Whizzer. In response to a Navy request for an experimental 5.0-in. motor to give the highest possible velocity, the 5.0-in. Motor Model 38 was designed. It was nicknamed the "White Whizzer" after the author's favorite football player, "Whizzer" White. The use of the motor was not originally specified, but it proved to be for the purpose of accelerating the ram jet motor which was being developed in the East at JAV–APL (Sec T). It was not felt desirable to use a longer grain than the Mk 18 unless absolutely necessary, and so the Mk 1 motor was simply lightened as much as possible. The motor tube was shortened by 5 in. and machined on the outside (except at the ends) to a wall thickness of 0.125 in., thus reducing its weight from 44.7 to 27.7 lb. The grid stool was lightened and shortened by eliminating the blowout disk, and some metal was removed from the nozzle plate to lighten it slightly. Suspension lugs were omitted and small lightweight fins, attached to a cylinder, were held in place by bolts into the nozzle plate. The result was a loaded motor which weighed 62.2 lb instead of the standard 88.3 lb. With the standard HVAR payload, its velocity was almost 50 per cent greater than the HVAR, and with light heads it was actually clocked at 2,490 fps. This velocity requires an acceleration in excess of $100g$, so that the force on the grain would certainly restrict the upper-temperature limit seriously, but no difficulty was found with it up to 100 F, which was the highest temperature at which it was tested. No information is available concerning the Navy's use of the motors which were supplied by CIT.

19.4.3 Launchers and Service Use

The launchers for the HVAR are the same as for the AR's except that its greater weight necessitated more rugged designs and impelled the change from Dural to high-tensile steel for post launchers, as mentioned in Chapter 17.[35,36]

After its first spectacular and successful test in Normandy, the HVAR was very little used by the Army because of failure to set up any adequate and comprehensive program of pilot training and failure to coordinate supply so that the rockets were available at the times and places where they might have been effectively employed. This situation was in the process of being remedied when World War II ended. With the Navy in the Pacific, the HVAR gradually supplanted the 5.0-in. AR as it became available. As anticipated, it proved to be a great improvement over the slower 5.0-in. AR, but the details of its use must be found in Navy publications.

19.4.4 Reports

On the ammunition itself, the two most important CIT reports are references 37 and 38. Various aspects of its use in forward firing are discussed in many of the reports listed in Section 19.2.7. Manufacturing problems are treated in references 39, 40, and 41. Motor-loading procedures, applicable essentially either to HVAR or "White Whizzer," are detailed in reference 42.

19.5 11.75-IN. AIRCRAFT ROCKETS

The much better accuracy and penetrating power achievable with forward-fired rockets than with bombs made desirable the development of an aircraft rocket which could carry a payload comparable to that of a large aircraft bomb. Sporadic tests of accelerating standard bombs with several small rocket motors had been made from time to time at CIT and elsewhere, but this was a clumsy and inefficient method of getting velocity and proved also to be very inaccurate. The obvious solution was one big rocket motor. Such a big motor became possible as soon as the 4.2-in. cruciform grain was available, and the development of the 11.75-in. aircraft rocket began in March 1944, soon after that of the HVAR. For obvious reasons, it was immediately nicknamed "Tiny Tim." The first field firing was made on April 26; one static firing of the propellant charge had been made two weeks earlier. The design was logically developed from the 5.0-in. HVAR and presented a number of problems not previously encountered in the project's work with smaller rockets. These included:

1. The use of a multiple-grain charge, which necessitated an internal structure for its support. Four Mk 19 cruciform grams, 60 in. long, were used, giving a propellant weight greater than total weight of a loaded HVAR.
2. The use of threads on the motor much larger than, and different in shape from, those in standard commercial use which can be made in ordinary machine shops with commonly available tools.
3. The requirement of special devices for handling and attaching these larger rockets to airplanes.
4. The large blast effect, which required (a) careful engineering to minimize, (b) special launching devices to separate the rocket from the airplane before ignition, and (c) a considerable program of research into the sighting and aiming problems of this type of launching.

The Navy's 500-lb SAP bomb AN–M58A1 appeared to be the most desirable head for such a rocket, and fortunately there was a standard oil well casing of the same diameter, 11.75 in. OD, which had adequate wall thickness and tensile strength and enabled the development program to get started without waiting for a special mill run of tubing. There was not much of it available, however, and we were reduced for a time to the expedient of salvaging it from abandoned oil wells.

Because of its size, which made production slow and posed extraordinary difficulties both in motor design and in installation on aircraft, the Tiny Tim was a long-term project in comparison to its predecessors. Nevertheless, its progress was very encouraging, and, when in June successful air firings began, it was decided that Tim was a likely supplement for the Holy Moses against the robot bomb launching sites. Thus on June 28, 1944, six days after the first air firing of Tiny Tim, a memorandum from the Navy Chief of Staff to the Vice Chief of Naval Operations assigned top priority to the development of the rocket and its associated launchers for the purpose of getting it into service as soon as possible. Work was to start immediately on prototyping launcher installations for the F4U and F6F aircraft, and the SB2C was later added to the list. Although the design of the internal motor components was not entirely settled, CIT undertook production of sufficient motors to be able to supply 10 per day to the Services. Several hectic weeks followed before the high priority was deferred on August 7 because it became clear that the bomb-launching sites would be captured before Tim could be put into action. Two weeks later the crash of an SB2C in an experimental test caused a sudden halt and a complete re-examination of the program, and in the ensuing months the difficulties with blast and the various internal ballistics problems were studied in detail and gradually worked out. Development was essentially complete by October 1, Navy contractors began setting up for production, and the rocket was ready for combat test. Minor design changes and refinements continued for several months thereafter, however, dictated for the most part by the requirements of fitting to various types of aircraft.

The following spring, aircraft squadrons with drop launchers were sent to the Pacific on the carriers *Franklin* and *Intrepid* for the first service test of Tiny Tim. The disastrous attack on the *Franklin* took place before its rocket planes ever went into action against the enemy, and the 500-lb explosive rocket heads in her hold contributed to her downfall. Although it is believed that the *Intrepid's* planes fired a few Tims against the Japanese, the Navy has not divulged any details. The Division 3 history says they were used on Okinawa.

Army Air Forces also undertook a program of outfitting appropriate planes for firing the 11.75-in. AR. This program would have had the planes ready for action in the final invasion of the Japanese homeland. The end of World War II left Tiny Tim as a potentially powerful and effective weapon, which would enable a plane to deliver the punch of a 12.0-in. gun, but a weapon which never had a combat test of its capabilities.

19.5.1 External Design Features

The original specifications called for a rocket to be fired from aircraft having a 500-lb payload and as high a velocity as possible (preferably at least 1,000 fps), and using as propellant four 4.2 x 1.5-in. cruciform ballistite charges. The rocket was to have multiple nozzles, including a blowout nozzle to increase its working temperature range, and for handling purposes it was to be capable of standing on its nozzle end. The first guess proved to be a good one on the two major components—the motor tube and the nozzle. Almost from the beginning their design was so stable that it was possible to continue regular production of them without consideration for the frequent and drastic revisions of internal design which occurred in the summer of 1944.

Motor Tube. The choice of propellant fixed the internal diameter of the motor tube as not less than, and preferably not much more than, 10⅞ in. Its wall thickness was determined by the specification that it stand a 4,800-psi internal pressure test without permanent yield. Since saving weight was a primary concern, it was desirable to use tubing of a relatively high yield point in order to keep the wall as thin as possible. The grade N–80 API oil well casing, with an external diameter of 11.75 in., a 0.489-in. wall thickness, and a minimum yield of 80,000 psi, was the most suitable material found; it had the additional advantage of having the same outside diameter as the 500-lb SAP bomb which was being considered as a possible high-explosive head for the rocket.

To obtain the required internal diameter it was necessary to machine the inside full length, and it was decided to machine the outside also, partly to save weight but primarily to assure accuracy. The 10 per cent permissible variation in wall thickness could displace the center of gravity of the motor tube from the geometrical center of the ID by as much as 0.3 in., but it was desirable to keep the overall mechanical malalignment of the rocket as small as the gas malalignment, which with multiple nozzles was expected to be less than 1 mil (0.06 in.).

The diameters chosen, 11.7 in. and 10.9 in., with a minimum wall thickness of 0.380 in., give a maximum fiber stress of 76,800 psi (calculated by Barlow's formula) for an internal pressure of 5,000 psi. It was realized that this wall thickness was probably ultraconservative, since it was based upon standards evolved by the project from experience with small motors which did not have a blowout disk to limit the maximum pressure in the motor. The fact that a burst of such a large motor would, it was believed, almost certainly result in destruction of the aircraft justified such conservatism, at least in the beginning. Later, tubing with a minimum yield of 90,000 psi became available and was specified by the Bureau of Ordnance for its production. Two high-temperature firings of Navy production motor tubes with walls reduced to 0.280 in. were successful, and for the final production design (the Model 5 motor) a nominal wall thickness of 0.340 in. was specified. From the performance standpoint, considerably more drastic reductions could be made, as was further shown by later tests at NOTS, Inyokern, of motors with 0.200-in. walls. The increased velocity which can be gained by reduction below 0.340 in. is not very great, however, and for combat use from aircraft it is believed that a thinner wall is not desirable in view of its increased vulnerability to gunfire.

The two ends of the tube were threaded internally, one to take the body and the other to take the nozzle. In order to get as much strength at the threads as possible, the outside machine cut was stopped about 3 in. short of the ends. The thread, a modified buttress with a 3-degree loaded face, a 50-degree included angle, and a pitch of 2½, was designed for maximum strength against internal pressures combined with ease of assembly. The choice of 3 degrees was rather arbitrary; it was desired to keep the angle small in order to minimize the tendency of the end thrust on the nozzle to expand the motor threads, and 3 degrees was one of the common standard angles for buttress threads. When the prime contractors for large-scale production began making inquiries about the design, it became evident that the choice had not been the best one, since the smaller the angle, the smaller the diameter of a thread grinding wheel or hob which can cut the thread. From this point of view, an angle of about 7 degrees would have been preferable, since it allows the use of tools 3 to 3.5 in. in diameter and is still less than the angle of repose for friction between slightly lubricated steel surfaces so that there would be no tendency to expand the tube. By this time, however, production of bodies with this thread was already under way by the Naval Gun Factory, and the Bureau of Ordnance has not considered the change desirable.

A glance at an early general arrangement drawing of the 11.75-in. aircraft rocket reveals that there was 6 in. of empty space at the head end of the Mod 0 motor. This came about as a result of a variety of factors. The original design of the structural members holding the propellant grains was such that it was expected that a considerable fraction of the gases would move forward and reverse their direction at the front end of the motor, and adequate space was necessary to allow this to occur without excessive heating. The original head design had a 12-in. overlap of the motor tube over the head for extra strength against oblique water impacts; the closure at the base of the body was a dome, convex forward, in order to leave the required space and still have the guiding land on the body as far back as possible. When the 11.75-in. Head Mk 1 was designed by the Bureau of Ordnance, the 12-in. overlap was reduced to 6 in. Because the design of the internal parts of the motor was so uncertain, it was decided not to reduce the length of the motor tube correspondingly.

Subsequently, two factors appeared which made a reduction in length desirable: the interference between the tail of the rocket and the wing flaps on certain aircraft and the fact that the bomb elevators in a considerable number of aircraft carriers would not accommodate motors longer than 80 in., but required the use of other elevators for transferring the rockets from the magazines to the planes. At a conference in December 1944 with representatives of the Bureau of Ordnance, it was decided that the motor tube should immediately be shortened as much as possible without changing the service heads (Mk 1 and Mk 2) in order that the outside length of the motor shipping box could be kept under 80 in. Modification numbers were assigned for the shortened motor, and production of the Mod 2 was begun by CIT as soon as new tubes could be made. It was subsequently found that the buttress thread was strong enough to stand water impact even without any overlap of the motor tube over the head. Consequently, in the final design (Model 5) the "skirt" on the head was removed and the motor tube was made as short as possible.

As has already been pointed out, the Model 5 motor was later found to be too weak to withstand ground impact and cannot be given a long underground trajectory even with the sphere-ogive head. Whether this is caused by the lack of overlap of the motor tube over the head, by the much thinner wall, or by a combination of the two factors has not been determined. Whether the Model 5 motor is actually an improvement on the previous designs, then, depends upon the tactical use. For most purposes, its higher velocity recommends it.

Nozzle Plate. In view of the success of the HVAR, the choice of a multiple nozzle (see Figure 11) for Tim was obvious. Various numbers of nozzles ranging from 6 to 80 were considered. The choice fell upon 24 as a good compromise between a large number of small holes, which could be made with a simpler tooling, and a small number of large holes, which would be cheaper. The holes were arranged 16 in the outer row, 8 in the inner row, and one large one in the center, which is closed by a heat-insulated copper disk unless the pressure during firing rises high enough to eject it and thus increase the nozzle port area. Since the motor was to be capable of standing on end, the usual method of bringing the electrical lead out of one of the nozzles could not be used. Special electrical receptacles placed in the nozzle plate were therefore designed. A special electrical cable is supplied with each motor to go between the receptacle on the aircraft and those in the nozzle plate.

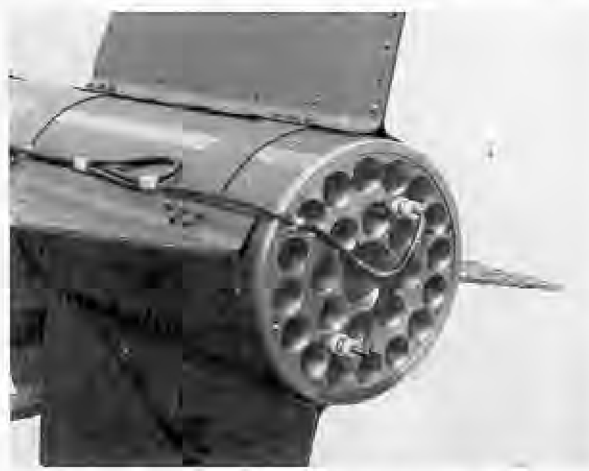

FIGURE 11. Nozzle end of Tiny Tim.

[redacted]

Tails. Considerable evolution took place on the tails, but nothing need be said about the early designs because they were all based on the idea of welding lugs onto the motor tube. Unfortunately, the author, who was responsible for the design details, was not a good metallurgist, and it proved to be absolutely impossible to weld even so much as a 7/16-in. stud on the N–80 motor tubing without getting occasional failures in the pressure test of the tubing or in field firing.

The tail which became standard was not of very elegant design and was never intended to become permanent, but Navy production began with it and World War II ended before the later improved design could be put into production. The individual fin pieces were made from 1/8-in. aluminum sheet, 24ST, with radial beads rolled into the metal 1/8 in. high for stiffening. The two halves were riveted together and to two steel bands which clamped on the motor tube, the rear band seating back against the ridge at the rear of the motor tube. The choice of aluminum over steel was made partly from weight considerations but chiefly because it was thought that less damage would be done to the propeller should a fin by any mishap get into its arc. The early fins were 12 in. wide, but in order to fit into the TBF bomb bay it was necessary to reduce them to 10 x 24 in., which became the standard. Interference with the wing flaps, which occurred with the adoption of the drop launcher in January 1945, caused the rear corner of the fins to be cut off, but, even with the corner removed, it was necessary on the Mod 0 motor to move the tail forward from its normal position in order to clear the flaps on some aircraft.

For the Model 5 motor an entirely new tail was designed. It was considerably lighter than the standard and had individually attachable fins so that they could be shipped in the motor box with a consequent saving of about 10 cu ft of storage space per motor. Since the Model 5 motor was not produced by the Navy, very few of the new tails were made, and still better designs have since been worked out at Inyokern.

With regard to fin shape, the conclusions of the water tunnel tests on the HVAR are probably equally valid for the 11.75-in. AR, and considerably wider fins would be desirable if they would fit on the aircraft. Tests of telescopic fins have been tried at NOTS, Inyokern, and such fins might significantly increase the accuracy.

Lug Bands. Lugs for attaching the motor to the airplane were originally welded to the motor tube, but this scheme had to be abandoned along with the welding of the fins, and the lugs also were attached to bands. This arrangement proved to be necessary for another reason, however, for it is impossible to use the same lug position on all aircraft. The Mod 0 motor was issued with the lug bands placed as required for the displacement launcher on the F4U, which was to have been the first installation to get into combat. Five bands were required: a standard bomb-hoisting lug at the center of gravity of the loaded round, two standard bomb suspension lugs to fit the standard bomb racks, and two launching lugs to attach to the displacement launcher and release the rocket at the bottom of the swing. In the latter part of 1944, tests on the drop launcher were so successful that the displacement launcher was declared obsolete and was removed from the airplanes. The drop launcher required only the three standard bomb lugs, but a second hoisting lug was attached at the center of gravity of the loaded motor for handling it before the head was attached. The change in lug band arrangement was made almost simultaneously with the change in motor tube length, so that almost all the Mod 0 motors had 5 bands, while almost all the Mod 2 motors and all Model 5 motors had 4 bands.

None of the lug bands made by CIT would stand up under the loads specified by the Bureau of Aeronautics, corresponding to accelerations of 13.4g vertically (i.e., radially) and 11.4g fore-and-aft. They were adequately strong for ordinary use, however, and until the internal ballistics problems were resolved, there was no time to worry about lug bands. When comprehensive tests were made, it became apparent that it would not be possible to make suspension bands out of ordinary cold-rolled steel that would be strong enough to prevent slipping or distortion under the specified loads without a considerable increase in thickness over the 3/16 in. that had been used. The bands being made by CIT would take, on the average, only about half the specified loads, and those from the Navy contractor would take even less. Even the bomb suspension lugs themselves were too weak. Consequently, it appeared desirable to adopt heat-treated 4130 steel for the whole assembly and thus obtain parts about which no question of strength would exist. A minimum yield point of 100,000 psi was specified,

[redacted]

and as this was two and one-half times the average of the cold-rolled $\frac{3}{16}$-in. bands, it was possible to reduce the thickness to $\frac{1}{8}$ in. and still increase the strength well above that required. Tests showed that $\frac{1}{8}$-in. suspension bands properly heat-treated would stand the vertical load test with a considerable margin of safety and could be tightened on the tube so securely (75 ft-lb torque on $\frac{1}{2}$-in. bolts) that either band alone would withstand the specified vertical and fore-and-aft loads, although in actual practice the loads would always be divided almost equally between the two lugs. These bands were recommended for Bureau production.

19.5.2 Internal Design Features

Blowout Disk. The central nozzle is closed by a shallow copper cup, clamped in place by a threaded retainer. The cup (usually called a disk) is insulated from the motor gases by a $\frac{1}{4}$-in. asbestos-filled bakelite plug and a layer of hard-setting Permatex. Originally the disk was 0.064 in. thick and sheared at a hydraulic pressure (cold) of 3,000 psi. It was found that this disk did not always blow out at 130 F, and, when it did not, high pressure peaks and much lower gas velocities were obtained in field firing. A reduction to 0.050-in. thickness, giving 2,250 psi as the cold shearing pressure, raised the average gas velocity at 130 F from 5,430 to 6,340 fps. Static-firing tests gave 3,120 ± 150 and 2,490 ± 115 as the actual mean blowout pressures of the two thicknesses of disks, slightly higher than, but in reasonable agreement with, the values obtained with cold water pressure. The later adoption of JPN in place of JP propellant,[b] with the consequently lower pressure at high temperature, brought a further reduction of the disk thickness to 0.043 in. in order to keep the safety factor of the motor as high as possible.

Grid. The grid design was fairly obvious and has caused no difficulty except that it was originally designed much heavier than proved to be necessary. In trimming down the Model 5 motor to the minimum in weight, about 10 lb was saved by supporting the grid on four legs instead of a ring. The ring was originally used to prevent erosion of the motor tube at the front face of the nozzle plate where the gases are deflected to go through the holes. This erosion had been found to be serious in the HVAR at high temperatures, but on the 11.75 in. it proved to be very small because of the difference in gas flow through the larger number of nozzles. To make doubly certain, the length of the motor tube threads at the nozzle end was made less in order to expose a minimum number of threads to the gases in front of the nozzle plate. That this change now made the two ends of the tube different was not thought to be a serious objection in large-scale production.

Structure for Mounting Propellant Charge. When the idea of using a multiple-grain charge was advanced, enough experience had been gained on smaller grains, particularly the 2.74-in. cruciform, to indicate that they would not burn stably and smoothly unless each grain was shielded from the radiation given off by the others [c] and fairly well supported mechanically along its whole length. It was also desirable that the grains be held firmly down against the grid even under backward accelerations of $12g$. The most persistent and difficult design problems arose in connection with the structure for accomplishing these ends.

Charge Support. Although, strictly speaking, nearly every internal part is a support for the charge, the name has been given to the structure which attaches to the grid at the rear end and serves to hide the grains from each other, supports them along their length, and attaches at the front end to the clamp which prevents the grains from moving forward. Only major variations in the charge support will be discussed here, since small changes were almost innumerable.

The earliest tests, with charge supports which completely surrounded each grain, were unsuccessful because such supports had to be made out of fairly thin steel (0.075 in. was used) in order to fit into the tube. Flight of the rocket was satisfactory, but virtually the entire rear end of the charge support was eroded away by the time the burning was three-quarters complete so that the grain broke up early and gave low gas velocity.

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[b] The original ballistite composition used by CIT (standard trench mortar propellant was designated JP for "jet propulsion." In 1944 a slightly different composition became standard and was designated JPN (N for "new"). An experimental composition designed to have higher strength was called JPH (H for "hard"). All compositions contain roughly $\frac{1}{2}$ nitrocellulose and $^3/_7$ nitroglycerin with small amounts of other compounds.

[c] Recent research at Inyokern has shown that radiation effects are actually not serious in this case, however, so that considerably lighter and simpler designs of charge support can be made. See Figure 12C.

[redacted]

The first successful charge support was the so-called "X type" shown in Figure 12A. It was formed from $\frac{3}{16}$-in. steel sheet and welded to the grid. It was realized that, touching the grain only at the corners, it might not give sufficient support, but it was simpler to make than other types which had been suggested, and the initial experimental tests were successful. It is probable that powder having a compressive strength as high as that of JPH would perform about as well in this charge support as in any other. When, however, a large quantity of ballistite was received with too high a nitroglycerin content and a consequently lower compressive strength, trouble was immediately encountered in high-temperature proof firing. The new powder gave high pressure peaks and excessive powder breakup, and on one round an effective gas velocity of only 4,620 fps was obtained. In the belief that the difficulty was probably insufficient mechanical support of the grain, tests were begun with a new charge support, the 4Y type shown in Figure 12B.

The success of the 4Y type in eliminating the bad high-temperature performance with 44 per cent nitroglycerin JPN propellant was spectacular. In one field test, it increased the gas velocity at 130 F by more than 1,200 fps and completely eliminated the end breakup peak as far as could be ascertained from the photographic data. The dimensional tolerances as originally laid down would have given the grains the same amount of support that they have in the 5.0-in. motor, in which the ends of all four arms are supported and the spacing between supports on opposite arms is very closely 4.625 in. This is accomplished in one direction by holding the arms of the Y's accurately and in the other direction by holding the size and concentricity of the central square section so that the spacing between it and the ID of the motor tube is correct. It was never possible to meet these close tolerances in the fabrication of the charge support, and the drawing tolerances were progressively loosened to be in accord with the facts. In ordinary service, apparently, a very loose fit of the grain in the support is adequate. With powder of low quality or in high-temperature firings, one would expect the gas velocity and the number of failures to depend on the snugness of the fit. Therefore, the author has always taken the attitude (in discussions with Navy contractors) that it is worth a little extra trouble and expense to make the charge support as accurate as possible,

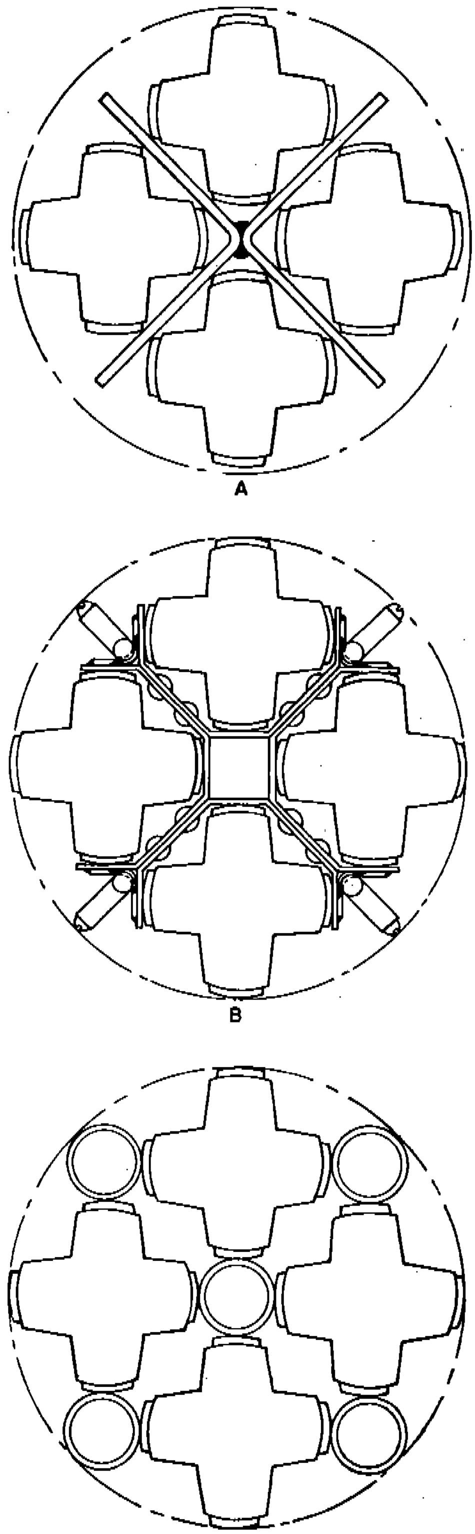

FIGURE 12. Charge supports for 11.75-in. motor. *Top:* X type. *Middle:* 4Y type now standard. *Bottom:* Tubular type which may supplant 4Y type.

[redacted]

even though it is impossible to prove experimentally that to do so will improve performance.

The 4Y was welded to the grid and had four threaded studs at the front for holding the charge clamp. It was made from 11-gauge steel in four sections which were originally spot-welded together, but, after some came apart during firing, a riveted design was tried and found successful. Elimination of the large amount of welding made it much easier to keep the parts straight and true.

For Navy contractors who preferred them, however, both spot-welded and arc-welded designs were included in Navy drawings as alternates. Also permitted was a design in which the charge support was bolted rather than welded to the grid, permitting the use of shims to get it concentric and coaxial with the grid. The arms of the Y's at the nozzle end showed a tendency to warp away from the grains during firing, and they were reinforced by pieces of 1-in. angle iron. The only other change was to shorten the support at the time of moving the igniters from the rear to the front of the charge clamp which is discussed later.

In the Model 5 motor an attempt was made to dispense with as much weight as possible. Various schemes were tried to make 4Y charge supports from 14- or 12-gauge material without success. It appears that nothing thinner than the standard 11-gauge material will hold its shape during firing well enough to give the grain its necessary support.

Charge Clamp. The charge clamp is bolted in place as one of the last operations in motor loading to hold the grains firmly against the grid. In order to accommodate small length changes in the grains, about 1½-in. thickness of cruciform felt washers is placed between the clamp and the grains. Originally the igniters were placed between the felt and the front end of the grain, but, with the igniter size then in use, this arrangement was found to subject the grains and the clamp to rather large forces upon ignition, and the igniters were then mounted on the front face of the charge clamp. Various less rugged designs of charge clamp were tried and found to be too weak to withstand the ignition forces. The final design was a ⅜-in. thick steel plate, torch-cut into the approximate shape of the four cruciform grains and bolted to four studs welded to the charge support (see Figure 13). With the igniter in front of the charge clamp instead of between it and the grains, the clamp can probably be made thinner and lighter if it should appear desirable.

Igniters. The early experimental motors used either 16 or 24 Mk 9 igniters, which were developed for the 3.25-in. motor and contained 35 g of black powder each. As soon as it was available, the plastic case igniter which was developed for the 5.0-in. Motor Mk 1 was adopted. This is 4.6 in. in diameter and 2.1 in. high, and has a powder compartment holding up to 200 g of powder and a wiring compartment for connecting the two electric squibs. Four of these igniters were used, one in front of each grain, giving a total of 800 g of black powder.

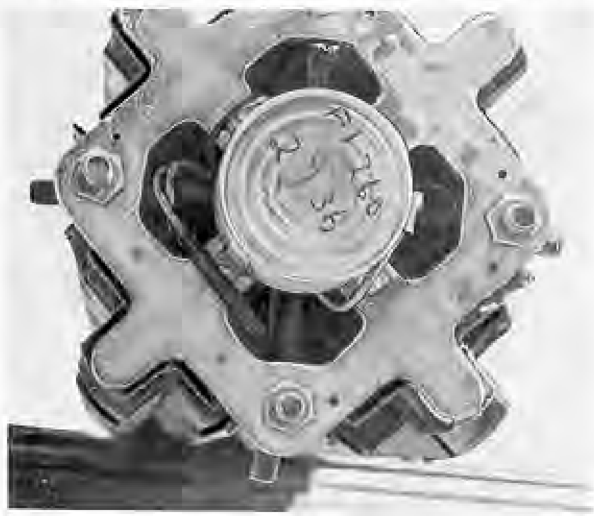

FIGURE 13. Front end of charge for 11.75-in. motor showing charge clamp and 230-g igniter. Later design eliminated exposed igniter wires.

In the early firings, in order to get the grains to temperature it was necessary to remove the nozzle charge support assembly from the motor. As a result, in low-temperature firings, considerable frost formed on the surface of the powder grains. One static test showed an igniter peak of only 400 psi and a 50-millisecond delay in reaching the equilibrium pressure of 850. Since in aircraft rockets it is desirable to have the pressure rise as rapidly as possible, it had been the general policy in determining the adequacy of an igniter to have a pressure peak at the low-temperature limit about as high as the equilibrium pressure. On the basis of the −35 F static performance it was decided to try a total of 1,200 g of igniter, by filling the wiring compartment as well as the powder compartment and drilling holes between them. In the initial static tests the increased igniter appeared satisfactory from −50 to 144 F, and it was adopted as standard. Later the

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four plastic igniters were superseded by four tin plate igniters of the same capacity, also containing two squibs. From the beginning it was intended that the tin plate igniters should be used when available, but the abandonment of plastic igniters was accelerated by the discovery of a piece of plastic which had been blown into the oil cooler of an F4U-1D, incapacitating the aircraft. The new igniters were made of 0.010-in. tin plate on ordinary tin can machinery with top and bottom crimped to the sides with the standard "double crimp."

In experimental firings from wing launchers on the SB2C airplane, either with fixed or "lanyard drop" launchers, there was severe damage to the elevators. Investigation with high-speed cinematography disclosed that the elevators were given a severe and brief acceleration, presumably by a shock wave, before the main propellant blast was set up. It was soon found that the magnitude of this shock wave is roughly proportional to the size of the igniter. Accordingly, it was decided that the igniter should be as small as possible even at the sacrifice of low-temperature performance, and a single tin can containing 230 g of black powder was adopted as standard.

In retrospect, it is clear that, if the grains had not been frost-covered on the early cold shots, we would not have concluded that 1,200 g of igniter was necessary. The proper amount from the standpoint of good ignition is probably 800 g or somewhat less. When this factor is balanced against the shock wave damage to the aircraft, it is very difficult to determine the optimum amount to use. Tests conducted at NOTS in March 1945 on the effect of igniter size on blast damage showed that the main blast was larger than the igniter blast up to about 500 g of igniter. It was therefore recommended that the igniter charge be doubled in the interest of better ignition at low temperatures. No such igniters had been made by the time the rocket was turned over to NOTS. As an alternative, two of the smaller igniters could be used, but this seemed undesirable since it increased the power requirements and complicated the design.

Igniter Leads. The method of connection and protection of the wires running from one or more of the igniters to the electrical receptacles in the nozzle plate was a persistent problem. Various troubles involved in making connection to four igniters will not be discussed. When the single igniter was introduced, the wires, which had formerly been brought out near the outside of the tin can, were moved to the center and a 1.0-in. hole was bored in the center of the charge clamp to admit them into the central square in the 4Y charge support. At the grid, the wires passed out of the central square through two rubber grommets (later combined into a single two-legged grommet) and thence to the receptacles. This arrangement was satisfactory except that the wires (about 10 ft of No. 16 stranded copper, insulated) were always ejected during burning. In an attempt to keep them inside, a number of schemes were tried: wrapping the wires around the grid, tying them to a rivet at the front end, running them through small holes in a bulkhead at the front end of the square, and plugging the central square with a plastic material which was cast around the wires. The design of the Mk 1 motor was frozen with no method of imprisoning the igniter wires. In the Model 5 motor, the wires were brought through the grid through small, snug-fitting holes without grommets. With this arrangement, almost all of the igniter leads remained in the motor during firing.

Occasional motors were found to be short-circuited because small flakes of steel and beads of weld dropped from the cracks in the charge support into the receptacle holes in the nozzle. To prevent this, the holes were filled with a plastic material. Several were tried, the best being "3–M Weather Strip Cement" (Minnesota Mining and Manufacturing Co.).

The electrical leads from the nozzle plate to the aircraft also caused considerable trouble, particularly in drop launching, because the wind force tended to break them and because they had to be coiled so as not to tangle. In the final design (shown in Figure 11) the joint between the two-conductor cable and the two single-conductor cables, at which breakage usually occurred, was eliminated by unraveling about 1 ft of the two-conductor cable, tying the individual insulated conductors in an electrician's knot, stretching them into the form of a T, and molding rubber over them. Numerous schemes for coiling the lead were tested and rejected. The method finally adopted was to lay the excess cable along the motor tube in one long loop and attach the loop by means of special aluminum clips to the length of cable running from the nozzle plate to the suspension lug at the center of gravity of the round. This design materially reduced the number of misfires in drop launching.

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Motor Seals. The design of primary and auxiliary seals for the motor was relatively straightforward, based on experience with the 5.0-in. HVAR. Each nozzle is sealed individually with a die-formed steel cup 0.010 in. thick. The seal for shipping purposes on the Model 5 motor is a shallow steel pan screwed against a rubber gasket on the rear face of the nozzle plate. An attempt was made to reduce the mass of the 24 little sealing cups, but it appears that thinner cups do not give a reliable seal (0.005 in. being entirely too fragile) and aluminum cups are destroyed in a short time by electrochemical action in a salty atmosphere. The front seal is a very tight-fitting steel dome inserted with a hydraulic jack, and a light disk in the thread protector gives further protection.

Heads. The first "service" head, which was hurriedly designed and put into production by the Naval Gun Factory when the high-priority service test was in prospect, was the 11.75-in. Rocket Head Mk 1. It was admittedly a stopgap and was made by welding a heavy adapter to the rear of a standard Navy 500-lb SAP bomb and machining the buttress threads on the adapter. It allowed only a single base fuze and was not properly sealed against the motor pressure. Later the Mk 2 was designed having essentially the same shape as the Mk 1 but a solid nose (the Mk 1 had a small nose fuze hole) and was forged in one piece. It was closed at the base by a steel plug accommodating three PIR or DDR base fuzes, and both the plug and the fuzes were sealed in place with gas check rings. It became the standard production head. For the Model 5 motor, the long "skirt" was removed from the Mk 2 head, making it the Mk 4. A sphere-ogive head (see Figure 13 of Chapter 24) was also designed and tested both under water and under ground. Although it had a much longer underwater and underground trajectory than the Mk 1 or Mk 2 heads, the latter were also stable, and so the sphere-ogive head was not put into production. Most CIT tests were made using practice heads. They consisted of a piece of tubing closed at the front with a standard dome-shaped welding head and are shown in several of the photographs.

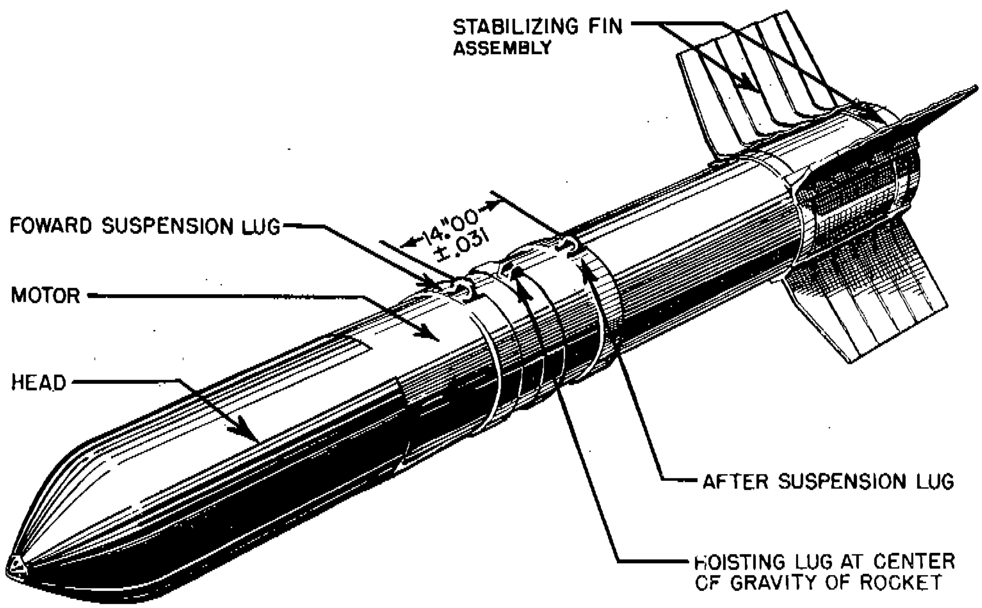

FIGURE 14. 11.75-in. rocket ready for loading on drop launcher.

Fuzes. Tim started out with the Mk 157 base fuze (Mods 1 and 2) because it was available and later used the improved Mk 163. The DDR fuze for Tim is designated Mk 162.

19.5.3 Types and Designations

The original motor (tube length 82.0 in.) was designated Mk 1 Mod 0 in CIT production and Mk 1 Mod 1 in BuOrd production. Mk 1 Mods 2 and 3 were assigned to the slightly shortened version (motor tube 75.75 in. long), and Navy production was changed to this design. Both rounds are designated Mk 3 with either Mk 1 or Mk 2 heads, but CIT nomenclature distinguishes between the long Model 3 and the "medium-short" Model 4. For the so-called "ultra-short" motor, the designation Mk 2 was assigned to Navy production motors, but none were ever produced, and motor and round usually go by the CIT name, Model 5.

19.5.4 Launchers

Aircraft launchers for Tiny Tim (see Figure 14) have been discussed in Chapter 17. Several ground-firing launchers were also built for proof-firing the rounds and for possible use against caves. Most of the launchers were of the two-rail variety, the guide consisting of two long parallel pipes supported so that one fin rides between them. Launchers of this type discussed in *Rocket Launchers for Surface Use* [43] are the Type 53 proof-firing launcher, the Type 55A launcher mounted on a two-wheel trailer, and the Type 59 "portable" launcher which sits on the ground on its own legs and can be carried by eight men. All these launchers are bulky (12 to 15 ft long) and cumbersome and, as in the case of the HVAR, do not apparently increase the accuracy over that of a much shorter launcher. A "zero-length" ground launcher, the Type 61 (see Figure 15), was therefore designed in which the front end of the motor is supported on a rotating sector and both ends became free simultaneously after 10 in. of motion.

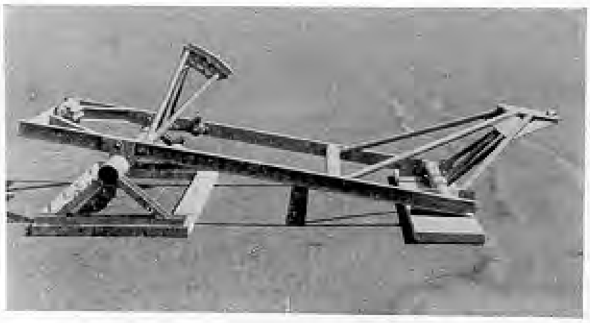

FIGURE 15. Experimental Type 61 "zero-length" ground launcher for Tiny Tim.

19.5.5 Reports

A full discussion of the design and development problems of the rocket motor is given in reference 44 from which much of the previous discussion is taken. The state of the ammunition at the time of the first high-priority program, with the X-type charge support and the Mk 1 head, is shown in a BuOrd pamphlet,[45] and later revisions show the production model. On proposed service uses, the only CIT reports are references 46 and 47. Later reports have all been put out by the Naval Ordnance Test Station, Inyokern. An illustrated article on drop launching is contained in reference 48. Manufacturing and inspection problems are discussed in reference 49.

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Chapter 20

SERVICE DESIGNS OF SPIN-STABILIZED ROCKETS

By *C. W. Snyder*

20.1 3.5-IN. SPIN-STABILIZED ROCKETS [SSR]

During the first two years of the project, all CIT's work was with fin-stabilized rockets. In this we were following the lead of the British, but it was undoubtedly a wise choice for fin-stabilized rockets involved fewer and generally simpler problems than spinners and could therefore be developed and put into service use more quickly. German rockets, however, were almost all spin-stabilized, and their success (especially against our Flying Fortresses) coupled with the hope of obtaining greater accuracy and more compact projectiles led to the initiation in 1943 of intensive research on spinners by both major rocket groups in this country.

At CIT a few rounds of experimental 4.5-in. spinning *barrage rockets* [BR] had been fired by the "Accuracy Committee" in the spring of 1943, but the first successful firing of a finless rocket was on the following October 13. This rocket, designated the 3R1 (i.e., 3.0-in., Rotating, Type 1), consisted of a standard 20-lb 3.5-in. Mk 1 head (solid steel antisubmarine *aircraft rocket* [AR] head), a 3.25-in. motor tube, and a nozzle plate held in place by a 3.5-in. diameter threaded ring. The eight nozzles, each with a 0.250-in. throat diameter, were canted tangentially at a 16-degree angle to give right-hand spin. Overall, the round had a length of 22.5 in. (6.4 calibers) and a weight of 29.75 lb. The 2.5-lb cruciform grain, seated on a "button" grid, imparted a velocity of approximately 550 fps. On the first test, both integral (i.e., bored out of a solid nozzle plate) and insert nozzles were tried, and, since both were satisfactory, the insert design was chosen. Since an explosive head was required, the Mk 1 head was quickly replaced by the 3.5-in. Head Mk 3, having the same weight but somewhat greater length. It was discovered that the dispersion could be significantly decreased by machining the outside of the head to a slightly smaller diameter (2.45 in.) except for about an inch at the rear, so that the launcher contacted the rocket only at the two "bourrelets," one formed by the rear portion of the head and the other by the nozzle ring. This rocket, fired seven weeks after the first one, appears at a casual glance almost identical with the one finally designed and standardized, but actually the development was only beginning.

In the ensuing months, such problems as the following had to be investigated. What are the optimum nozzle cant angle and the maximum quadrant elevation for stable flight, and how do these affect one another? Where should be the center of mass and what should be the shape of the nose to give minimum dispersion or maximum quadrant angle? How long can the rocket be and how fast can it go and still remain stable? How does dispersion vary with launcher length, and what is the effect of malalignment, of dynamic unbalance, of tip-off, and of wind? To discover the answers to many of these questions took more than a year and a very considerable number of rounds.

The original exploratory work on spinners took more definite form as the result of a request by the Marine Corps for a spinner which might be substituted for the 75-mm pack howitzer. For this application, a tubular launcher mounted on a .30-caliber machine gun tripod was developed, the final model being the CIT Type 42B or Mk 40 Mod 0. In comparison with the pack howitzer, the rocket and this launcher had a considerable advantage in lighter weight and consequently greater mobility, but, because of its higher dispersion, the rocket was not adopted for service use. Various other possible uses of the rocket were suggested at different times and launchers for them were tested, but by the end of World War II no 3.5-in. spinners had been sent abroad.

20.1.1 Design Features

Grain. The 2.74-in. cruciform shape was chosen for the initial tests because of its ready availability and because it was felt that, since its inhibitor strips would remain in contact with the motor walls throughout burning, it would be less subject to being fractured or thrown off center by the spin forces. Inhibitors were cemented to both ends of the grain and to the rear half of the outer ends of all four arms. This 50 per cent inhibiting would normally give a progressive burning curve, but with the very small nozzles erosion is so severe that the actual burning curves are quite regressive at high temperatures and slightly so at medium temperatures. Although the cruciform shape, having been

FIGURE 1. 3.5-in. spinner components.

originally designed for much longer grains (Mk 13) gave a very low loading density (internal K only 35), it fulfilled the requirements and was never changed. It was designated Mk 23. A few early tests were made with 5.0-lb cruciform grains, but, because of the greater length and higher velocity, these rockets were not stable.

During the winter of 1944 to 1945 two other grain shapes were tested in standard motors. One was a 2.5-lb "hexaform" (six-legged) grain, which gave performance almost identical with that of the standard except at low temperatures where it was superior to the standard. The other was a 3.09-lb tubular grain, which also performed satisfactorily, giving velocities above 950 fps as compared to the standard 750 fps.

Igniter. Brass can igniters containing 20 g of powder were used originally, but they ignited the grain rather slowly so that the low-temperature performance was not good. Thirty-five-gram plastic case igniters were then tried, but after a thorough test it was found that plastic case igniters gave 50 per cent greater dispersion than brass can igniters, the explanation presumably being that pieces of cellulose acetate were plugging some of the nozzles, at least temporarily. The final solution to the igniter problem, in this as in most other cases, was a metal case igniter, the Mk 18 Mod 0 containing 30 g of powder. In this as in all spinner igniters, the "false crimp" (see Figure 14 of Chapter 22) was used to reduce the impact on the propellant grain caused by the bursting of the closely confined igniter case.

Grid. Strictly speaking, the 3.5-in. SSR does not have a grid, but the term has always been used to denote the little button on which the grain sits. As previously mentioned, button grids came very nearly being satisfactory even for the 8.5-lb Mk 13 grain, so they were the obvious choice for the much shorter spinners. The original grids were 2 in. high, but it was quickly shown that a reduction in height even to ½ in. gave no significant change in performance, and this dimension became standard.

Nozzle Plate and Ring. Eight nozzles were originally chosen for symmetry with respect to the cruciform grain. Six nozzles were found to give no increase in dispersion and were preferable from the production standpoint as well as giving less erosion because of their larger size. As previously mentioned, insert nozzles were chosen both for lightness and for cheapness, and furnace brazing was found to be the most satisfactory method of holding them in the nozzle plate. Any method which holds them securely is apparently equally good. To assure accurate alignment and secure fastening, very close tolerances were found to be required on the nozzles and on the holes in the nozzle plate; the former were centerless-ground to an outside diameter of $0.812 + 0.001 - 0.000$ in. and the holes were reamed to $0.812 + 0.000 - 0.002$ in.

The first guess on nozzle cant angle was 16 degrees, and, although this rather large angle probably improved the performance of the early rather long spinners which were fired at low angles, a slower spin was required for good high-angle flight. A cant angle of 12 degrees proved to be the optimum not only for the 3.5-in. spinner but for all the 5.0-in. barrage spinners as well.

Since the rockets were to be used from automatic

[redacted]

launchers, contact rings were required, and the design finally evolved was the same as that for the 5.0-in. spinners (shown in Figure 4). The "hot" contact ring was molded into a bakelite insulator which slipped over the rear skirt on the nozzle ring, which itself formed the ground terminal. Rivets through insulated grommets held the contact ring in place and made electrical contact to the igniter lead inside the nozzle ring. To prevent ignition failures it was found desirable to solder the current-carrying rivet to the outer contact ring.

Since the motor tube had external threads, the nozzle plate seated on the end of it, and proper nozzle alignment could be obtained by checking the alignment with respect to the front surface of the nozzle plate and checking for squareness of the end of the motor tube.

Heads and Motor Tubes. The first spinner to be fired, using the button grid 2 in. high, had a motor tube 13⅝ in. long and a 3.5-in. Mk 1 head, making the overall length approximately 25 in. Substitution of the Mk 3 HE head increased the overall length by almost 5 in., and, although this rocket had sufficient spin to be stable in spite of its length (the cant angle was still 16 degrees), it would not follow a 45-degree trajectory unless the conical nose was replaced by an ogive of 4 calibers radius or more (8 calibers was usually used). The reason for the superiority of the long ogive was that it moved the center of pressure forward relative to the center of mass, thus increasing the overturning moment so that it could cause the rocket to follow the turning trajectory without exceeding the permissible yaw.[a] With only half as much spin (8-degree cant angle) the rocket performed well at both low and high angles with the conical nose, but with an 8-caliber ogive nose was unstable at all quadrant angles because the stability factor was too low.

Reduction of the length of the grid button by 1½ in. gave a motor tube 12⅛ in. long, and the length of the head was successively reduced so that the payload dropped from 20 to 18½ and then to 14½ lb. The 8-caliber ogive continued to be popular for experimental rounds, but, when the question of a suitable nose fuze arose, it proved to be simpler to use the conical Mk 100 without changing its exterior contour. Proper igniter design made possible a further reduction in motor length leaving a minimum of space at the front end. The final motor tube had a length of 11½ in. and had a light skin-cut machined on the exterior to reduce variations in wall thickness and consequent unbalance.

Seals. Motor seals, both front and rear, were identical, except for size, with those for the 5.0-in. spinners (see Figure 3 of this chapter and Figure 13 of Chapter 23), but the nozzle end seal was changed to that shown in Figure 14E of Chapter 23 so that the extending edges of the seal would hold the round in place in the tubular aluminum launcher Type 37D, which, at the time World War II ended, was expected to go into service use.

Fuzes. Various nose fuzes were used in the course of development of the 3.5-in. spinner, but all were relatively minor modifications of the Army M48 fuze, as is explained in detail in *Rocket Fuzes*.[1a] This design of fuze was chosen because it was found that the feature of optional *delay* or *superquick* detonation was very effective with the rocket. Tests showed that with the fuze set *superquick*, ground craters were about 1 ft deep and 3 ft in diameter; with the fuze set *delay*, the rounds either ricocheted giving airbursts with a good fragment pattern 20 to 30 ft wide at low impact angles, or dug in at high impact angles making craters 3 ft deep and 4 ft in diameter in hard ground. It would also penetrate and detonate behind about 8 ft of sandbags, 3 ft of logs, ⅝ in. of mild steel, or more than 1 ft of concrete at normal incidence.

20.1.2 Designation and Types

Only one model of 3.5-in. spinner was standardized and recommended for service use. It was the 3.5-in. Rocket Mk 5 Mod 0, designated by CIT as the 3.5-in./4 Model 24A. The /4 GPSR means "approximately 4-thousand-yards-range General Purpose Spinning Rocket," and the model number alone is a sufficient designation. It consists of the 3.25-in. Motor Mk 13 Mod 0 (CIT Model 6), the 3.5-in. Head Mk 13 Mod 0 (CIT Model 12), and the Nose Fuze Mk 100 Mod 0 with Auxiliary Detonating Fuze Mk 44 Mod 2. For rounds with inert-filled heads, the practice was to use a model number 100 greater than that of the explosive-loaded round, so that the standard round (inert) is designated Model 124.

[a] The dynamics of spinners and how the yaw causes it to keep aligned with the trajectory is explained in Sections 21.5.1 and 25.5.

[redacted]

20.1.3 Launchers

The original service launcher designed for Marine use in place of the pack howitzer was the Mk 40 Mod 0 shown in Figure 2. A very large variety of guide shapes was tried in an effort to get one which would be easy to manufacture and give minimum

FIGURE 2. Two views of Mk 40 launcher for 3.5-in. spinner.

dispersion. No single design appeared definitely superior to all others, but that of the Mk 40, 3 ft long and tubular with three internal guide rails, was as good as any and was adopted as standard for most spinner launchers, both 3.5-in. and 5.0-in.

A number of other launchers for various purposes were tested, including two varieties of automatic launcher: a light, smooth bore, aluminum tube launcher for use in jungle warfare and sabotage, and a closed-breech launcher for replacing the 37-mm gun on the LVT-A1 armored amphibian tractor.

20.2 5.0-IN. SPIN-STABILIZED ROCKETS

The last CIT rockets which saw large-scale action in World War II were the 5.0-in. spinners. The initial test of such a rocket was on January 3, 1944. Its primary purpose was to provide the PT boats with a heavy, high-velocity weapon of sufficient accuracy for use against the armored and armed barges which the Japanese were using for supplying their island garrisons. It will be recalled that, late in 1943, the Commander Motor Torpedo Boat Squadrons had begun equipping his PT's with launchers for the 4.5-in. BR and had had good success with them, but the rocket's comparatively low velocity and large dispersion made it far from ideal.

It was hoped that a velocity of at least 1,600 fps could be achieved with a 20-lb payload. The first tests were made with a 12-lb charge consisting of four tubular grains. Overall length of the rocket was 37.2 in. or approximately 7.4 calibers, slightly shorter relatively than the 3.5-in. spinner which was current at the time. Its head was the front part of a 5.0-in. Mk 1 (5.0-in. AR head). Rounds were fired at spin velocities of approximately 100, 200, and 300 rps with various nose shapes and weight distributions, and none would fly stably to the end of burning. The variety of combinations tried was great enough to make it reasonably certain that a rocket of that length could not be stabilized without a considerable increase in spin velocity, and rounds with 400-rps spins burst at the end of burning because of the centrifugal force. One group of rounds fired with a reduced charge to give a velocity of approximately 1,000 fps instead of more than 1,600 fps was just on the verge of instability, two out of three rounds flying stably. It thus appeared that it was the increase in overturning moment associated with the supersonic velocity that was causing the trouble.

An attempt was then made to shorten the round and lighten it as much as possible so that approximately the same velocity could be obtained with a shorter and lighter grain. This rocket, with a length of 5.76 calibers and a velocity of more than 1,500 fps flew perfectly and gave, on its initial firing, a dispersion at low quadrant angle of only 4 mils. At the same time a 4.2-in. cruciform grain was substituted for the multiple-grain charge because of the poor static performance of the latter. Subsequent tests showed that the length could be somewhat increased, and 6.3 calibers was adopted. This rocket, with the addition of a Mk 100 fuze, a metal case igniter, and the necessary electrical contact system became the 5.0-in./10 GPSR Model 20 and ultimately the 5.0-in. Rocket Mk 7 Mod 0.

As soon as it had been shown that 5.0-in. spin-stabilized rockets up to about 6½ calibers would fly and indeed would give considerably better dispersions than were attainable with fin-stabilized rockets, applications for them multiplied rapidly. In particular, the Navy was interested in a rocket which would supplement the 4.5-in. barrage rocket and have a longer range, since offshore obstacles such as reefs sometimes kept the rocket-firing boats too far away from the beachhead to accomplish their purpose. (This was the case, for example, during part of the Saipan operation in June 1944.) A 5.0-in. spinner seemed to offer the best possibility for this application, since ranges even up to 10,000 yd were easily obtained and their shape made them easily adaptable to automatic launching.

During the summer of 1944, various other models of 5.0-in. spinners appeared, having either the full 10.1-lb propellant grain of the Model 20 (later designated the Mk 21 grain) or one half as heavy (the Mk 22). Then in the fall the Navy drew up plans for a rocket gunboat which was to utilize the full potentialities of the spinners. The Bureau of Ordnance was to develop a continuously reloadable launcher (the Mk 102) with remotely controlled adjustable elevation and train, and the gunboat, which was to use the LSM hull, was to be designed especially for mounting ten of these new launchers together with four mortars, one 5.0-in. gun, and various automatic weapons.

Also as part of the plan, CIT began an integrated development program on barrage spinners which was to produce rockets with three different ranges—5,000, 2,500, and 1,250 yd—all having the same weight (about 50 lb) and the same length so that they would all fit the same launchers and could be handled and stored in the same manner. For each range, a variety of heads would be available:

1. *Common* [Cn]. Semi-armor-piercing, with explosive D loading and a base fuze.

2. *General purpose* [GP]. A moderately thick-wall shell (about ½ in.) with TNT loading and nose fuze.

3. *High-capacity* [HC]. A thin-wall shell (about ¼ in.) with maximum TNT loading and nose fuze.

4. *Smoke* [Sm]. A very light-wall shell (about ⅛ in. thick) with either WP or FS filling, a nose fuze, and a tetryl burster.

5. *Chemical warfare* [CW]. Similar to the smoke head but designed for filling with chemical agents of lower density (1.43 or less).

6. *Pyrotechnic* [Py]. A light-wall shell with time fuze and separating charge to eject an illuminating flare and parachute combination.

This ambitious program was far from complete by the end of World War II because, in contrast to the case for finners, where the principal consideration in fitting a motor to a head is the thread size, the necessity for keeping weight and length constant and still getting a maximum payload for each rocket meant that every new design was a completely new problem. Out of the total of eighteen possibilities, six were completed, and one, the 5.0-in./5 HCSR Model 34, was given a round Mark number (Mk 10 Mod 0) and put into extensive service use.

In October 1944, experiments in forward-firing spinners from aircraft were begun. As might have been expected, the very large wind forces to which a rocket launched in this manner is subjected before it reaches its maximum spin velocity made necessary still shorter rockets and higher spin velocities than had been satisfactory for ground firing. The development of a satisfactory forward-firing round required a considerable amount of research, both experimental and theoretical and in particular involving the solar yaw camera. More of the details of this research are given in *Firing of Rockets from Aircraft*,[2] and in *Field Testing of Rockets*.[3] By the fall of 1945 when the problem was turned over to NOTS, Inyokern, the 5.0-in./14 GASR Model 39A, having a 19-lb payload and a velocity of 1,330 fps, had been developed to the point where its accuracy was as good as the best fin-stabilized aircraft rocket and the general problems of aircraft spinner ballistics were fairly well understood.

20.2.1 Spinner Designations

The number of 5.0-in. spinner combinations which existed, at least on paper, was more than thirty, and it would serve no useful purpose to list them all. Each combination was distinguished by a round model number, but to make the terminology more descriptive it became customary to include in the designation the general type of the round, using the abbreviations given for the six types listed in

[redacted]

the preceding section, and the approximate range in thousands of yards for ground firing at 45 de-degrees QE. Thus "5.0-in./10 GPSR Model 20" signifies that the round has a "general purpose" head and a range of approximately 10,000 yd. Actually its range turned out to be greater than expected, 10,880 yd, but the designation was not changed. The exception to the general rule is the 5.0-in./14 GASR Model 39 (*general purpose aircraft rocket*), where the 14 signifies approximate velocity in hundreds of feet per second, since the round is too stable to follow a 45-degree ground-fired trajectory. The two models of 5.0-in. Rocket Mk 7 were formerly called *high-velocity spin-stabilized rockets* [HVSR].

20.2.2 5.0-in. Rockets Mk 7 [HVSR]

In discussing the design of the various spinners, it will be convenient to take first the 5.0-in. Rockets Mk 7 Mods 0 and 1, the two high-velocity spinners for PT boats, and later to point out the changes

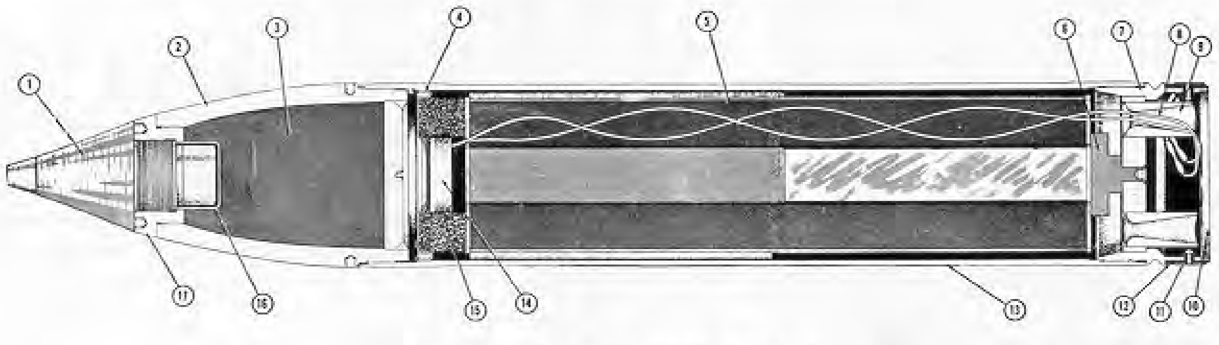

FIGURE 3. 5.0-in./10 GPSR Model 20.

1. Fuze (Mk 100)
2. Head (Mk 7)
3. Filler (TNT)
4. Front seal
5. Propellant (Mk 21)
6. Button grid
7. Nozzle ring
8. Igniter leads
9. Nozzle
10. Rear seal (obsolete design)
11. Contact ring
12. Insulating bushing
13. Motor tube
14. Igniter (Mk 17)
15. Felt washer
16. Fuze liner containing Mk 44 Mod 2 auxiliary detonating fuze
17. Fuze liner ring

which were necessary to adapt the basic design to the other models. The two rockets have the following components:

5.0-in. Rocket Mk 7 Mod 0 (5.0-in./10 GPSR Model 20);
Head Mk 7 Mod 0 (Model 8) with Mk 100 nose fuze;
Motor Mk 3 Mod 0 (Model 4).

5.0-in. Rocket Mk 7 Mod 1 (5.0-in./9 CnSR Model 32);
Head Mk 8 Mod 0 (Model 30) with Mk 31 base fuze;
Motor Mk 3 Mod 0 (Model 9).

Grain. The Mk 21 Mod 0 grain has a length of 16.2 in. and weighs 10.1 lb. It is inhibited with four 8¼-in. long inhibitor strips. These were originally placed two at the rear on opposite arms of the cruciform grain and two at the front on the other arms. Static tests showed no change in performance when other patterns—all four strips at the front, center, or rear; or right- and left-hand spiral patterns—were used, but high-temperature field firings with patterns having no strips at the rear (i.e., all four strips at the center or the front) showed a considerably increased tendency for motor bursts. Since spiral patterns were no better, the original pattern was made standard because it was somewhat simpler. Both ends of the grain are also inhibited, of course.

Igniter. As in the case of the 3.5-in. spinner, some difficulty was experienced with the closely confined igniters. The 55-g Mk 14 (HVAR) igniter, for example, fractured the grain badly in partial burning tests at −10 F. A smaller false-crimped metal igniter, on the other hand, performed satisfactorily with a minimum of free space at the front end of the motor. The igniter adopted, Mk 17

Mod 0, is the same one used in the 3.5-in. spinner [b] and is held in a hole in the center of a 1-in. thick felt ring, the hole being eccentric so that the igniter leads can come out of the case into the space between two arms of the grain.

Motor Tube. On the basis of experience with the 3.5-in. spinner that best accuracy was obtained with two points of contact with the launcher, two bourrelets were machined on the motor tube. The NE 8735 HVAR tubing was used, which ran considerably over its nominal 3/16-in. wall thickness. To lighten it, as well as to reduce variations in wall thickness which might introduce dynamic unbalance, the tubing was machined to 4.937 ± 0.005-in. outside diameter except near the ends where the bourrelets were left $4.970 + 0.000 - 0.010$ in.

Nozzle Plate. The nozzle end design is shown in Figure 4. A height of 7/8 in. for the button grid was chosen on the basis of static tests as the shortest that gave no change in the pressure-time curves. Eight nozzles were chosen because this number gave a convenient size for machining and gave an expansion ratio of 4 with somewhat less length than six nozzles. The choice of 12-degree cant angle was relatively arbitrary and a somewhat larger angle might have been preferable for the flat trajectories in which the rocket is used, but the choice was made to give stable flight at 50 degrees QE. The electrical contact system is virtually identical with that of the 3.5-in. spinner. The V-shaped groove just ahead of the contact ring accommodates a spring latch to hold the round in place in launchers such as the trailer-mounted Type 44 or the Type 49B PT-boat launcher (see Figure 8).

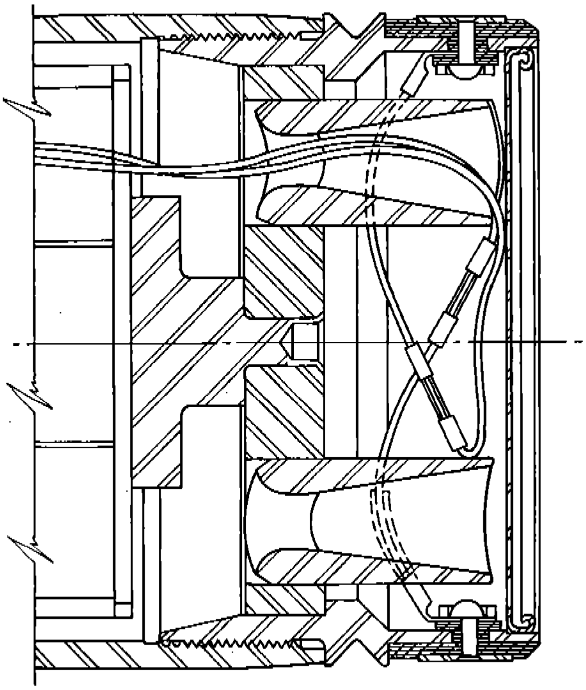

FIGURE 4. Details of nozzle end of 5.0-in. spinner motors with cruciform grain.

At the request of the Bureau of Ordnance, the skirt on the nozzle ring was for a time made considerably thicker than is shown in Figure 4 (0.273 in. instead of 0.093 in.) because it was felt that the thin skirt would not stand the forces to which the continuously reloadable Mk 102 launcher would subject it. It was later found that such was not the case, and the thin nozzle rings again became standard.

Heads. The "general purpose" head Mk 7 (see Figure 7) was made by cutting off the rear 9.75 in. of the Mk 1 head, welding in a 1/2-in. thick steel plate as the base closure, and threading. It was intended to weigh 20.0 lb with the Mk 149 nose fuze, having been originally designed for the *high-velocity aircraft rocket* [HVAR] but never used with it except for experimental tests. With the nose fuze Mk 100 Mod 0 and the auxiliary detonating fuze Mk 44 Mod 2, the head weight is almost 1 lb less. Against unarmored or lightly armored targets, this head works very well. For example, in impact at 45-degree obliquity with 5/8-in. STS armor, fuze set *superquick*, it tears a hole 2 ft in diameter. With the fuze set *delay*, high-order detonation after penetration of 1/2-in. mild steel plate was observed at 0-degree and 30-degree obliquity. Its more rugged construction was the principal factor in the choice of the Mk 100 fuze over the T–28, which would not stand impact with 1/2-in. plate.

The alternate Mk 8 head was designed for use against somewhat heavier armor. It uses a standard Mk 31 projectile base fuze. Having no hole in the nose and being made from heat-treated NE 8744 steel, it functions properly against 1-in. STS armor at up to 45-degree obliquity. On heavier plate or at higher obliquities, the head broke up but the fuze functioned. The velocity of the rocket is great enough that it will punch out a disk from 1 1/2-in. STS even though the head deforms badly and breaks. It is thus clear that a still more rugged head is justified and highly desirable for this rocket.

[b] The Mark number is different because of the different lead length.

[redacted]

20.2.3 High-Capacity Spinners [HCSR]

The "high-capacity" series was the only one of the six proposed series of 5.0-in. barrage spinners which CIT completed. Its members are

5.0-in./5 HCSR Model 34 (5.0-in. Rocket Mk 10 Mod 0);
  Motor Mk 4 Mod 0 or Mod 2 (Model 6);
  Head Mk 10 Mod 0 (Model 38);
  Grain Mk 22, 5.6 lb, 9.1 in. long.
5.0-in./2 HCSR Model 51A;
  Motor Mk 5 Mod 2 (Model 51A);
  Head Mk 12 Mod 5 (Model 51);
  Grain Mk 24, 3.88 lb, 6.3 in. long.
5.0-in./1 HCSR Model 50D;
  Motor Mk 6 Mod 2 (Model 50B);
  Head Mk 13 Mod 0 (Model 50B);
  Grain Mk 25, 3.1 lb, 5.0 in. long.

The motors vary in length to fit the powder grain and accommodate as large as possible a payload, keeping the overall length 32.2 in. for all three, but otherwise their design is identical with that of the Mk 3 except in the following particulars.

Grids. For the 5.0-in./5 the same 7/8-in. high button was used, but for the two shorter ones the internal K is so extremely small that a 5/8-in. high button was found to work equally well.

Nozzle Plates. A cant angle of 12 degrees gave optimum high-angle flight for all three models. Eight nozzles were used in the 5.0-in./5, but with the very small propellant weights of the other two, four nozzles were sufficient and, of course, cheaper. As originally designed, the 5.0-in./5 was stable up to 65 degrees QE, the 5.0-in./2 up to 60 degrees, and the 5.0-in./1 only a little above 50 degrees. It was found that the addition of a 1½-lb weight to the nozzle plate, held in place by a longer stem on the grid button, increased the limit for the latter up to about 57 degrees.

Igniters. All three use the 30-g false-crimp metal case igniters, the designations being Mk 20 or Mk 18 according to the length of the wires.

Heads. The three heads are identical except for length, being made in three parts—rear closure, body, and fuze adapter—and silver-soldered together. To insure that the head does not extend radially beyond the bourrelets and strike the launcher guides, the body walls are made thicker than desired (4.95 in. OD) and machined to 4.89 in. OD after silver-soldering so that the exterior surface is concentric with the rear threads. The Mk 30 Mod 3 nose fuze was chosen for the HCSR series.

Also designed but not tested by the end of World War II was the 5.0-in./10 HCSR. By using a cylindrical grain with a higher loading density than the cruciform, the propellant weight could be increased to 9.8 lb and thus give approximately 10,000-yd maximum range to a payload about two-thirds that of the 5,000-yd rocket.

20.2.4 Smoke Spinners [SmSR] and Chemical Spinners [CWSR]

Of the SmSR and CWSR series, only the 5,000-yd models were completed. They are

5.0-in./5 SmSR Model 41A;
  Motor Mk 4 Mod 0 (Model 6);
  Head Model 54A;
  Grain Mk 22, 5.6-lb cruciform.
5.0-in./5 CWSR Model 61;
  Motor Model 61;
  Head Model 61;
  Grain 4.9-lb cylindrical three-ridge.

The former has the same motor as the 5.0-in./5 HCSR. The latter motor is designed after that of the 5.0-in./14 GASR Model 39, and the greater compactness of the tubular grain allows an increase in volume of the head filler by about 15 per cent over the former.

Head designs are similar to that of the HCSR heads except for the thinner wall and the addition of a tetryl burster extending almost the full length of the head. To keep the centrifugal force from displacing the long slender burster tube, it is supported at the rear by a spider and at the front by the fuze adapter.

20.2.5 Pyrotechnic Spinners [PySR]

Three PySR's were designed for three different illuminating flares, two having approximately 5,000-yd range and one approximately 4,000. The latter used the Mk 4 motor. None of them were tested thoroughly, but they appeared relatively satisfactory in preliminary trials. The 5.0-in./4 PySR Model 40 is described in *Ballistic Data*.[4] The CTSR time fuze was developed for them.[1]

20.2.6 Aircraft Spinners

Experiments in forward-firing spinners from aircraft began in the fall of 1944 using the 5.0-in./10

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GPSR, having a spin velocity of 250 rps and an overall length of 6.3 calibers. The results were highly unsatisfactory, the dispersion being very large because the rounds were unstable in flight. On impact they did not penetrate the ground, but flopped about, spinning rapidly, and in a few cases reaching a vertical position, nose down, spinning like a top. A record of the yaw in a plane perpendicular to the sun's rays, obtained by a solar yaw camera in the head of one of the rockets, is given in Figure 5, where it is apparent that the nutation amplitude built up to a very large value.

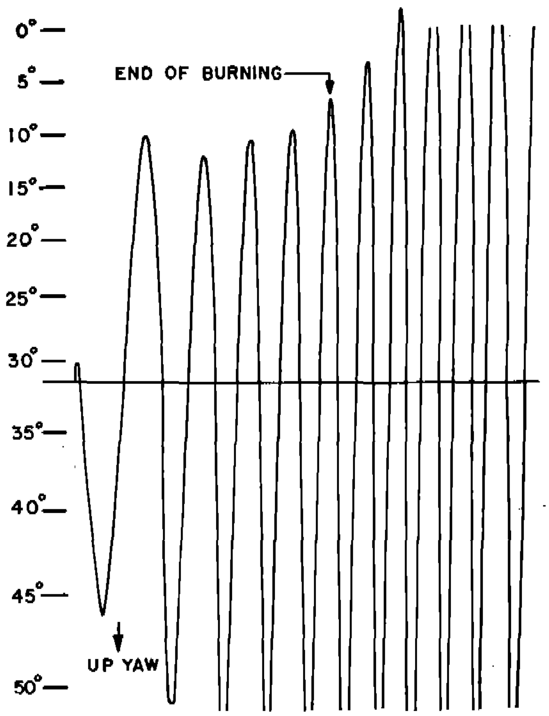

FIGURE 5. Yaw of 5.0-in./10 GPSR forward-fired from aircraft (taken from yaw camera record). Because of too great length, the rocket is unstable.

The first successful forward firing was done with a "hybrid" round consisting of the Mk 4 motor from the 5.0-in./5 HCSR and the Mk 7 head from the 5.0-in./10 GPSR. The shorter motor gave a spin of only about 150 rps, but the reduction of the length to 5.4 calibers made the round so much more stable in spite of it that the dispersion immediately dropped to about 8 mils and the yaw camera records began to look like that in Figure 6.

As a result of this success, a program of research on propellant grains was undertaken in an effort to increase the velocity of this round as much as possible. By eliminating the space both at the front and the rear of the grain to an absolute minimum, it was found possible to use a 10.1-in. length of 4.25 x 1.25-in. three-ridge tubular ballistite, weighing almost 7.9 lb. With this grain and a change in nozzle cant angle from 12 degrees to 16 degrees, the "hybrid" round became the 5.0-in./14 GASR Model 39A, shown disassembled in Figure 7. The only change in the motor was to remove the button grid and substitute in its place a ring grid, visible in the photograph, which seats in slots on the front face of the nozzle ring.

It was found that stability and dispersion were considerably better at higher spins, and the Model 39A has a maximum spin velocity of 310 rps. The large centrifugal forces which such spin velocities generate makes the propellant problem a difficult one, especially at low temperatures where the powder becomes brittle. The Model 39A is not considered safe below about 40 F. To remedy this difficulty, and also to increase the velocity still further if possible in the hope of making the GASR into an effective air-to-air weapon, research with internal-burning grains which fit snugly into the motor tube was begun by CIT and has been continued by NOTS, Inyokern.

20.2.7 Launchers and Service Use

The most important launchers developed for the 5.0-in. spinners, outside of BuOrd's Mk 102, with its capacity of 30 rockets per minute continuously, are the CIT Type 49B PT-boat launcher (Mk 50) and the CIT Type 52 automatic launcher (Mk 51), shown in Figures 8 and 9.

The Mk 50 launcher comes in two varieties: Mod 0 for starboard and Mod 1 for port. The units are mounted on the bow of the boat by means of a pedestal (not shown in the picture). They can be swung inboard for loading and outboard to allow the blast to clear the deck during firing. The elevation is adjustable, but the train is determined by aiming the boat itself. The rounds are fired in pairs. Several hundred launchers were built by CIT and BuOrd, and it is reported that they proved to be effective, but no detailed reports of specific actions have been made available.

The Mk 51 automatic is very similar to the Mk 7 automatic for the 4.5-in. BR, is intended for the

[redacted]

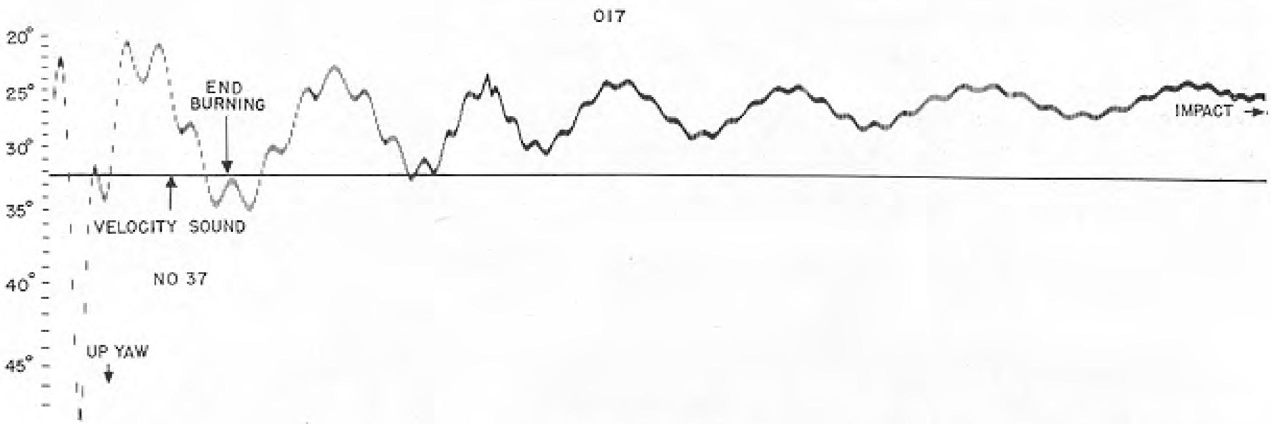

FIGURE 6. Actual yaw camera record of "hybrid" GASR forward-fired from aircraft. Increased stability resulting from short length causes large initial yaw to damp out rapidly.

FIGURE 7. Components of 5.0-in./14 GASR Model 39. Grid (third from left) fits in slots in nozzle ring (left).

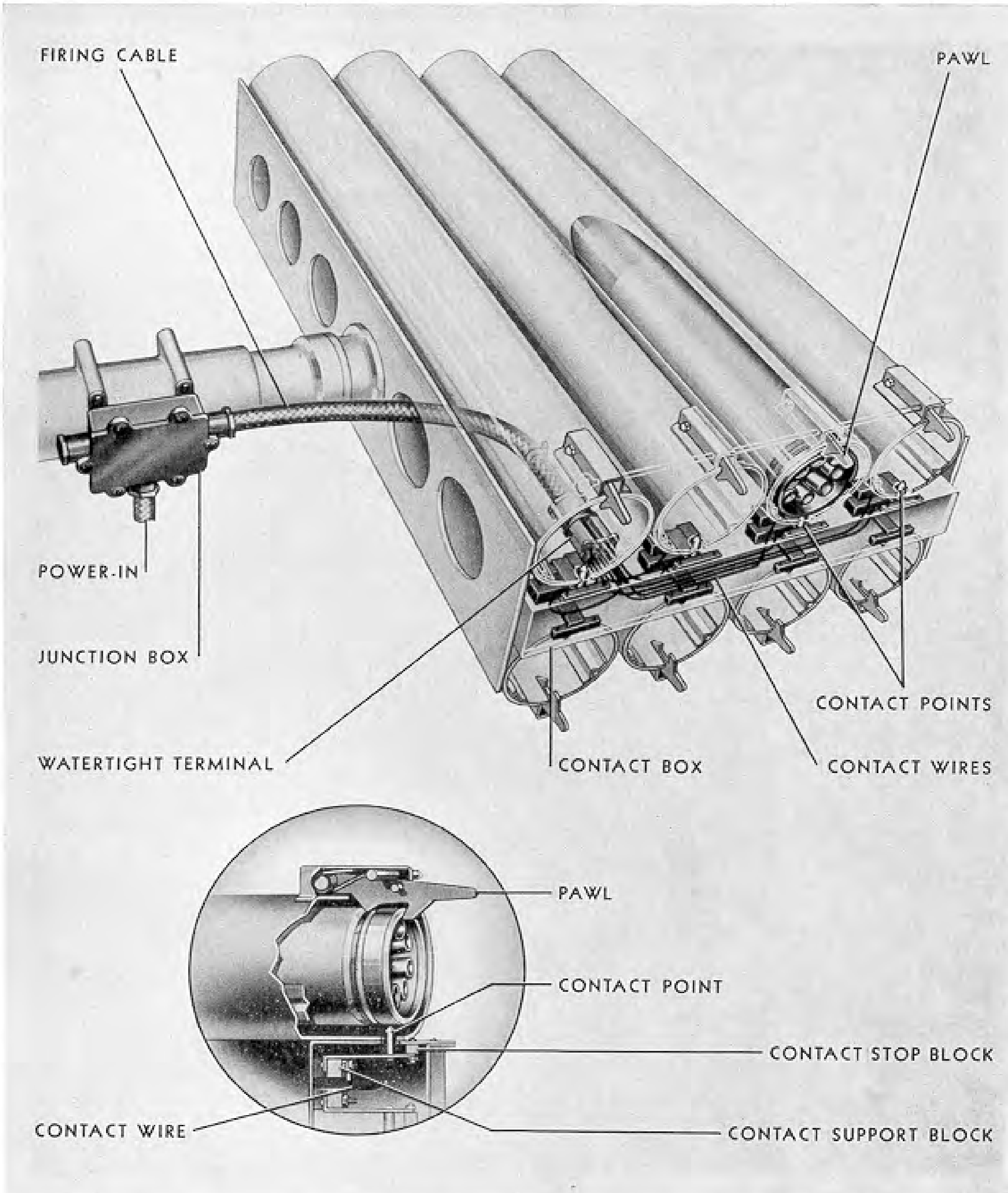

FIGURE 8. Launcher Mk 50 Mod 0 for PT boats, showing 5.0-in./9 CnSR in place.

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same purpose, and has the same universal applicability. They were used in the Pacific on four of a flotilla of twelve "interim LSM(R)'s," so-called because they were built to fill in until the "ultimate LSM(R)'s" with their ten Mk 102 launchers and other automatic weapons could be put into action.

FIGURE 9. Mk 51 automatic launcher with full load of 5.0-in./5 HCSR.

Each ship carried 85 automatics, making its total firepower more than a thousand rounds without reloading. The other eight ships were equipped with rail launchers for the 5.0-in. AR because the production of spinners was not yet great enough to supply the contemplated very large-scale use of rockets. The flotilla went into action on March 26, 1945, and continued in operation through June 15. The Okinawa operations probably represented the most varied and extensive use of rocket gunboats during World War II, and the spinners carried their full share of the load. In addition to bombarding the beaches themselves, the rockets were used to neutralize towns, knock out roads and railways, and fire away at whatever targets the aircraft observation spotters assigned to them. The troop command to which the four "spinner" ships were assigned reported that the effectiveness of the rocket fire was excellent at all times.

One other use of spinners deserves to be chronicled, although its effect on the outcome of World War II is hardly measurable. One submarine commander attached the base plate of a Mk 51 launcher permanently to the topside of his craft and stored a launcher and a supply of rockets below. When the submarine surfaced at an appropriate distance from the target, the launcher was brought out, attached to the deck frame, and loaded—the whole process taking about three minutes. In three different attacks, this submarine fired a total of 72 rockets in shore bombardment of the island of Honshu, reporting that all the rounds fell within the target area. Though it may not have helped much, it surely was fun.

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PART V

ROCKET ORDNANCE: THEORY, PRINCIPLES, AND DESIGN

By *E. B. Bradford*

THE INTRODUCTION by C. W. Snyder to Part IV applies equally to Part V. In Part IV he has reviewed solid-fuel rockets, their components and their launchers, primarily from the point of view of their employment as weapons, with special emphasis on practice as exemplified in the rockets developed during World War II at the California Institute of Technology. The basic principles of rocket propulsion and the war-end status of the theory and practice covering rocket propellants and interior ballistics have been ably reviewed by B. H. Sage, R. E. Gibson, and F. T. McClure in Parts II and III.

In Part V Snyder explains in greater detail the principles underlying the design of rockets for efficient performance in flight. He reviews rocket ballistics rather thoroughly, covers its application to the design of rocket propellant charges and rocket motors, and surveys the applications to fin stabilization and spin stabilization. These chapters provide physical explanations for the rocket behavior and limitations cited in Part IV.

The general conclusions are, of course, all derivable from physical principles long known, but their applications to rockets, with precision enough to make possible the rapid and substantial improvements in performance, had to await the data, much of it obtained by especially devised instrumentation made available from the extensive programs of rocket design and testing during World War II.

The emphasis in Part V is technical rather than military.

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Chapter 21

GENERAL THEORY OF ROCKET PERFORMANCE

By *C. W. Snyder*

21.1 THE MECHANISM OF PROPULSION

The thrust force which propels a rocket is the reaction to the high-velocity rearward flow of propellant gas [a] out through one or more nozzles. The function of the propellant is to generate gas to maintain high pressure and rapid discharge over a period of time (the "burning time"—0.3 to 3 seconds for the rockets of interest here).

Table 1. Ballistic quantities for fin-stabilized rockets.

$\theta \equiv$	angle between the horizontal and the tangent to the trajectory at any time.
$\theta_0 \equiv$	quadrant angle of elevation; angle of the launcher above the horizontal; (degrees, radians, or mils).
$\sigma \equiv$	yaw oscillation distance; distance rocket travels while executing one complete oscillation cycle in its yaw; (ft).
$A_N \equiv$	cross-sectional area of nozzle throat (sq in.).
$C_D \equiv$	aerodynamic drag coefficient; see footnote *i*.
$C_N \equiv$	nozzle coefficient; ratio of thrust to product of nozzle pressure and nozzle throat area; (dimensionless).
$c \equiv$	deceleration coefficient; defined by equation (16); (ft^{-1}).
$d_b \equiv$	burning distance; distance measured along trajectory through which rocket moves while burning; (ft).
$F \equiv$	thrust; force exerted on the rocket by the action of the jet at any time; (lb).
$G \equiv$	acceleration of the rocket; (in units of g).
$g \equiv$	acceleration of gravity; approximately 32.2 ft/sec^2.
$M \equiv$	projectile mass; mass of the rocket without propellant; (slugs); the weight W in lb is more often used.
$m \equiv$	instantaneous propellant mass; mass of propellant grain at any time during burning; (slugs).
$m_0 \equiv$	initial propellant mass; mass of propellant grain before burning; (slugs); the weight w_0 in lb is more often used.
$P \equiv$	pressure in motor chamber (assuming no pressure gradient); (psi).
$P_N \equiv$	nozzle pressure; pressure in motor chamber measured just to the rear of the nozzle end of the grain ahead of the nozzle itself; (psi).
$t \equiv$	time (seconds).
$t_b \equiv$	burning time; see footnote *d*; (seconds).
$V \equiv$	velocity of the rocket at any time; (fps).
$V_0 \equiv$	corrected velocity of the rocket (sometimes called "initial velocity"); velocity at the end of burning assuming no gravity drop and no air drag; (fps).
$V_b \equiv$	burnt velocity; actual velocity of the rocket at the end of burning; (fps).
$V_g \equiv$	effective gas velocity relative to the nozzle; defined by equation (4) or (6); (fps).
$W \equiv$	projectile weight; weight of the rocket without propellant; (lb).
$w_0 \equiv$	propellant weight; weight of the grain before burning; (lb).
$X \equiv$	horizontal range assuming impact point and firing point at the same elevation; (ft).

[a] In accordance with the established practice of rocketeers, we shall, for brevity, refer to the product of combustion of the propellant as "the gas," even though it is a complex mixture of many different gases.

21.1.1 Momentum-Impulse-Thrust Relations

In accordance with Newton's laws of motion, the forward momentum of the rocket increases during any time interval by an amount equal in magnitude to the backward momentum imparted to the gas ejected. Using the notation of Table 1, this fact is expressed as follows. In an infinitesimal interval dt powder of mass dm is burned and, as gas, flows out the nozzle with an average effective velocity V_g relative to the nozzle. It is assumed that V_g is the same for all masses of gas. Relative to the earth the gas has velocity $V - V_g$ and hence momentum $(V - V_g)dm$. The rocket's momentum at any time is $(M + m)V$, and its change during the interval considered is

$$\frac{d}{dt}[(M + m)V]dt = (M + m)dV + Vdm. \quad (1)$$

Hence we have

$$(M + m)dV + Vdm = (V - V_g)dm; \quad (2)$$

$$-\frac{dm}{M + m} = \frac{dV}{V_g}. \quad (3)$$

By integrating (3) over the burning period, during which m changes from m_0 to 0 and V from 0 to V_0, we have

$$\frac{V_0}{V_g} = \ln\frac{M + m_0}{M} = \ln\frac{W + w_0}{W}. \quad (4)$$

A less accurate but more frequently used expression is obtained by considering that the average mass of the rocket during burning is $M + \frac{1}{2}m_0$ and

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setting equal the total momenta acquired by the rocket and the gas. Thus

$$(M + \tfrac{1}{2}m)V_0 = m_0 V_g, \tag{5}$$

or

$$\frac{V_0}{V_g} = \frac{m_0}{M + \frac{1}{2}m_0} = \frac{w_0}{W + \frac{1}{2}w_0}. \tag{6}$$

Note that in either equation (4) or (6) only the ratios of masses and velocities are involved, so that any convenient units may be used. Equation (6) is the basic equation of rocket external ballistics. If V_g is assumed to be known, it enables us to predict the velocity which, in the absence of gravity and air resistance, will be imparted to a rocket of given weight by a given amount of propellant. Actually, it is a definition of the "effective gas velocity" V_g, and is used to calculate that quantity from velocities of rockets measured in field firing. As we shall see, the value of V_g depends upon the propellant used, the design of the rocket, and the initial temperature of the propellant. For ballistite it is never far from 7,000 fps, and more accurate guesses can be made from experience with similar rockets, so that equation (6) can be used to predict to within perhaps 5 per cent the velocity attainable with a rocket of proposed design. The actual velocity of the rocket at the end of burning, denoted by V_b, will differ somewhat from V_0 because of the effects of air drag and gravity.

It should be noted that equation (6) is exact only if the ratio of propellant weight to rocket weight is very small. The error is 0.6 per cent or less for all rockets now in service,[b] but it becomes increasingly less accurate as the relative weight of the propellant is increased and must be replaced by equation (4). Thus, for a rocket consisting of 63 per cent propellant and 37 per cent metal parts, equation (4) gives $V_0 = V_g$ (so that the last bit of gas expelled before burning ceased emerges with zero velocity relative to the earth) whereas equation (6) gives V_0 almost 8 per cent too low.

Another method of evaluating V_g is to hold the rocket stationary and measure the force it exerts on its supports. The relation is obtained from another of Newton's laws which states that the force exerted on the gas (and hence its reaction on the rocket) is given by the rate of change of its momentum. The mass of gas outside the rocket is $m_0 - m$ and its momentum is $(m_0 - m)V_g$ at any instant. Hence

$$F = \frac{d}{dt}[(m_0 - m)V_g] = -V_g\frac{dm}{dt}. \tag{7}$$

By integration over the burning time we obtain

$$V_g = \frac{1}{m_0}\int_0^{t_b} F\,dt = \frac{g}{w_0}\int_0^{t_b} F\,dt. \tag{8}$$

In "static firing" the thrust is measured as a function of time and the "integrated thrust" or "impulse" $\int F\,dt$ is calculated from the record. The specific impulse [c] or impulse delivered per pound of propellant burned, is $(1/w_0)\int F\,dt$. Like V_g, this is a measure of the efficiency of the rocket motor. It is obviously desirable to have both as high as possible.

21.1.2 Burning Time[d] and Acceleration

The principal differences between the external ballistics of rockets and of other artillery result from the disparity in the times of acceleration. A rifle shell is accelerated only while it is in the bore, a time of the order of 0.01 second, whereas a rocket is accelerated as long as the propellant burns—roughly 1 second. As a result, the forces exerted on a shell during firing are roughly a hundred times greater than those experienced by rockets. Force being proportional to acceleration, the acceleration of a rocket is an important ballistic quantity. Its average value is determined approximately from the time of acceleration (customarily called "burning time," t_b) by the relation:[e]

$$G_{\text{avg}} = \frac{V_0}{g t_b}, \tag{9}$$

[b] Even for the 5.0-in. Rocket CIT Model 38, the "White Whizzer" (see Section 19.4.2), in which the propellant is a higher percentage of total weight than in any service rocket, the error is only 1.0 per cent.

[c] Specific impulse is customarily given in pound-seconds per pound. Effective gas velocity, thought of as efficiency, has dimensions poundal-seconds per pound, which is equivalent to velocity.

[d] Since the burning does not stop abruptly, it is necessary to adopt an arbitrary definition of the burning time in terms of the shape of the pressure-time curve. Various definitions have been used for various purposes, but we shall not be concerned in this book with the differences among them.

[e] Actually V_b rather than V_0 should be used in equation (9) (see Section 21.1.1), but the relation is useful only for order of magnitude anyhow.

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the result being given in units of g, the acceleration of gravity. For the types of rockets designed by CIT, the upper limit of permissible acceleration (set by propellant strength) has been found to be roughly $100g$. The burning time thus cannot in general be less than approximately 0.3 second per 1,000 fps of velocity.

Again assuming the acceleration to be a constant, we can calculate the "burning distance" to an accuracy sufficient for almost all purposes from the simple relation:

$$d_b = \tfrac{1}{2} V_b t_b. \tag{10}$$

21.1.3 Relation of Pressure to Thrust

If we inquire into the origin of the thrust F in equation (8), we enter the realm of interior ballistics. The burning of the propellant produces a large quantity of hot gases inside the motor chamber at an equilibrium pressure which will be denoted temporarily by P. Since this pressure pushes equally in all directions against the walls of the chamber, it would produce no resultant force except for the fact that, on the area A_N of the nozzle throat, it finds nothing to push against to balance the force on an equal area at the front end of the motor. The resultant force on the rocket is thus given as

$$F \approx PA_N. \tag{11}$$

This formula requires correction because of two phenomena which were not considered in the foregoing simple discussion. First, because of the impedance to the gas flow from the front to the rear of the motor chamber, a pressure gradient exists, and a more exact analysis will show that it is the nozzle end pressure P_N which must be used in the formula. Second, there is an additional force on the rocket most of which comes from the forward component of the pressure of the expanding gases in the nozzle exit cone. A quantitative explanation of this additional force involves thermodynamical considerations and is relatively complicated. For practical purposes, its effect is taken care of by introducing into equation (11) a proportionality factor C_N, the "nozzle coefficient" or "thrust coefficient," which is a function of the nozzle shape and the pressure. The value of the coefficient is known from the theory of supersonic jets and from experimental data and is plotted in Figure 1 as a function of the "expansion ratio," i.e., the ratio of nozzle exit area to throat area.

With these two corrections, equation (11) becomes

$$F = C_N P_N A_N. \tag{12}$$

We can now eliminate F between (8) and (12), obtaining (when $V = 0$)

$$V_g = \frac{g}{w_0} A_N C_N \int_0^{t_b} P_N dt \tag{13}$$

(see footnote c) which gives us still another relation for determining the effective gas velocity. By means of a "static-firing" apparatus,[f] a record of the

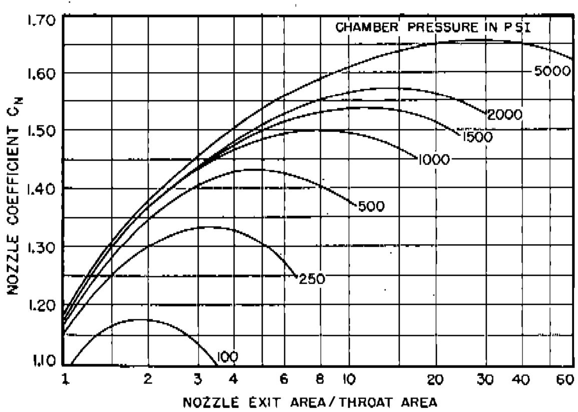

FIGURE 1. Theoretical values of nozzle coefficient.

nozzle end pressure as a function of time can be obtained. Measuring the area under the curve with a planimeter gives the value of the pressure-time integral, and the effective gas velocity can be calculated by equation (13) if the nozzle coefficient is assumed known. Alternatively, if one measures both pressure and thrust (as is usually done), an experimental value of the nozzle coefficient can be obtained by eliminating V_g between (8) and (13):

$$C_N = \frac{\int F dt}{A_N \int P_N dt}. \tag{14}$$

A typical pressure-time curve with the calculations on it is shown in Figure 2.

It was mentioned previously that the effective gas velocity is a measure of the efficiency of the rocket. Equation (13) shows that it is connected with interior ballistic constants through the factor

[f] Reports and equipment reflecting the static-firing experience of CIT are available at the Naval Ordnance Test Station, Inyokern, California.

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C_N which depends primarily on the nozzle shape so that, for evaluating the efficiency of a rocket propellant charge, a quantity more meaningful for interior ballistics is the ratio of effective gas velocity to nozzle coefficient.

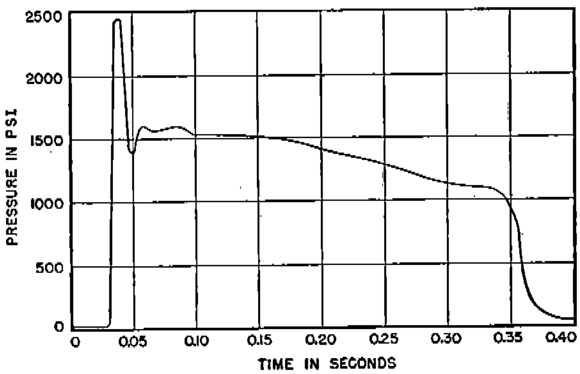

FIGURE 2. Typical pressure-time curve for ASR motor.

Area under curve = 18.02 sq in. (on the scale of the original record where the squares are 1 in.).
Conversion factor: 1 sq in. = 500×0.05 = 25 lb-sec per sq in.
$\int P_N dt = 18.02 \times 25 = 450.5 \approx 450$ (since the record is not accurate to better than 1 per cent).
Nozzle dimensions: Throat diameter = 0.781 in.
Throat area A_N = 0.479 sq in.
Exit diameter = 1.75 in.

$$\text{Expansion ratio} = \left(\frac{1.75}{0.781}\right)^2 = 5.0.$$

From Figure 1, for expansion ratio = 5.0, pressure = 1,500 psi, C_N = 1.50.
Propellant weight w_0 = 1.50 lb.

$$\text{From equation (13): Effective gas velocity } V_g = \frac{g}{w_0} A_N C_N \int P_N dt$$

$$= \frac{32.2}{1.50} \times 0.479 \times 1.5 \times 450$$

$$= 6{,}940 \text{ fps.}$$

$$\text{Impulse} = C_N A_N \int P_N dt = 1.5 \times 0.479 \times 450 = 324 \text{ lb-sec.}$$

Metal parts weight M = 61 lb.

From equation (6):

$$\text{Corrected velocity } V_0 = V_g \frac{m_0}{M + \frac{1}{2} m_0} = 6{,}940 \frac{1.50}{61.75} = 168.5 \text{ fps.}$$

21.1.4 Effect of Propellant Temperature

No mention of the variation of ballistic constants with temperature has yet been made. These variations are discussed in detail in Chapter 22. Like most other chemical reactions, the rate of burning of the propellant is faster at higher temperatures, and hence the burning time is shorter and the equilibrium pressure higher. Because less of the heat energy of the propellant is required for warming the rocket and more is available for pushing it, the effective gas velocity increases with increasing temperature over most of the temperature range. In some rockets, other factors enter at very high temperatures to reduce V_g again. In case temperature gradients exist within the rocket, the temperature of the surface of the propellant grain appears to be the controlling one.[1]

21.2 THE RANGE OF ROCKETS

21.2.1 Range in Vacuum

In the absence of air resistance, the range of a projectile in free flight is given by the well-known expression:

$$X = \frac{V_0^2 \sin 2\theta_0}{g}, \tag{15}$$

where θ_0 is the "quadrant elevation," the angle of projection measured upward from the horizontal. In this simple form, the expression gives the horizontal distance between two points on the trajectory at the same elevation. Thus for a rocket, if we use V_b instead of V_0 and the actual angle of the trajectory at the end of burning instead of θ_0, it gives the horizontal distance between the end of burning and the point on the downward trajectory at the same height. The total range is obviously greater than this by approximately twice the horizontal component of the burning distance. The correct expression is complicated because of the effects of tip-off at the launcher and because of gravity drop during burning. (See Chapter 24.)

21.2.2 Range in Air; Effect of Drag

For any but the very slowest rockets, the actual range is considerably less than the vacuum range because of the resistance of the air. The discrepancy is only about 3.5 per cent for the antisubmarine rocket,[g] but more than 45 per cent for the 5.0-in. HVAR.[h] The effect of air resistance is most easily

[g] We shall use the term *antisubmarine rocket* [ASR] for a group of rockets which are frequently called "Mousetrap ammunition." Although differing slightly in details, all these rockets were designated 7.2-in. Rocket Mk 1 Mod 0 in the latest Navy nomenclature. They have 2.25-in. motors and velocities of 175 fps or less. (See Figure 1 of Chapter 18.)

[h] The 5.0-in. *high-velocity aircraft rocket* [HVAR], often called "Holy Moses," has a velocity (in ground firing) of 1,360 fps and is the fastest fin-stabilized service rocket developed by CIT. It is shown in Figure 5 of Chapter 17.

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introduced by means of the "deceleration coefficient" c,[i] defined by the equation:

$$-\frac{dV}{dt} = cV^2. \tag{16}$$

For velocities up to approximately 800 fps, the resisting force offered by the air is very nearly proportional to the square of the rocket's airspeed, so that

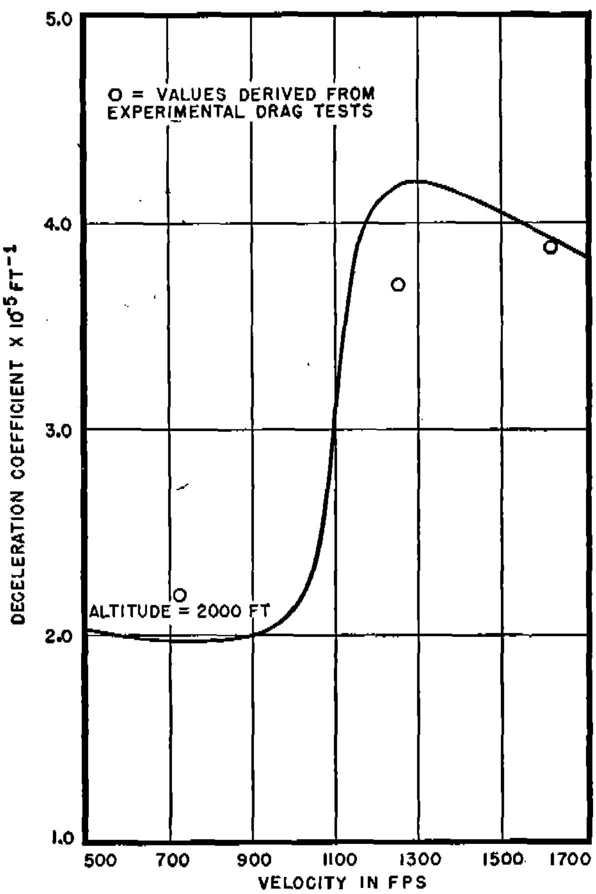

FIGURE 3. Deceleration coefficient of 5.0-in. HVAR.

c is a constant which can be fairly accurately estimated from theoretical considerations [2] or measured experimentally in a wind tunnel or water tunnel or by actual field firings. Its value for service rockets ranges between 1×10^{-5} and 9×10^{-5} ft^{-1}. A knowledge of the deceleration coefficient makes possible fairly accurate range calculations for rockets of subsonic velocities by the use of range tables for shells. Such calculations are discussed in Chapter 24.

When the velocity of a projectile begins to approach that of sound, the air drag becomes proportional to a higher power of the velocity than the second, so that, if we wish to continue to use equation (16) as its definition, the drag coefficient c must be considered a function of velocity. The exact form of this variation depends upon many factors, including the density, length-diameter ratio, smoothness, and nose shape of the projectile, and is not the same for a typical rocket as for a typical shell. Consequently, for high-velocity rockets, the accurate calculation of trajectories is very much more difficult and uncertain. A typical curve of deceleration coefficient vs velocity, that for the 5.0-in. HVAR, is shown in Figure 3.

For ground-fired rockets, the result of this variation of c is that, even though its burnt velocity is well above sonic velocity, the rocket quickly slows down to approximately 1,000 fps, so that attaining a range greater than that corresponding to the vacuum range for 1,000 fps (approximately 10,000 yd) is extremely difficult for short-burning-time rockets which are expected to carry a payload. This fact is illustrated in Figure 4 where approximate ranges are plotted as a function of initial velocity and deceleration coefficient.[j]

21.3 SPIN-STABILIZED ROCKETS [SSR]

The foregoing discussion has been written in terms of fin-stabilized rockets (usually called "finners" for brevity), but it is, for the most part, equally applicable to spin-stabilized rockets ("spinners"). Before considering the factors in which finners and spinners differ drastically from one another, we shall note the alterations which must be made in the equations of the preceding pages if they are to apply to spinners. Although rockets have been made to rotate by a variety of devices, including canting the fins,

[i] Also frequently used is the "aerodynamic drag coefficient" C_D, which is related to c by the formula $C_D = 2Wc/A\rho$ where W is the weight in pounds, c is the deceleration coefficient in feet^{-1}, A is the maximum cross-sectional area in square feet, and ρ is the density of the medium (air or water) in pounds per cubic foot. C_D is dimensionless.

[j] Figure 4 is based on reference 3 and assumes that c varies in the same way for all projectiles according to the Gâvre function (see Section 24.4.2). This approximation is fairly accurate for shells, which have little variation in the ratio of length to diameter and no fins or lugs to complicate the problem. Its accuracy for rockets can be estimated from the experimental points plotted. The value of deceleration coefficient quoted in each case is that for velocities well below sonic.

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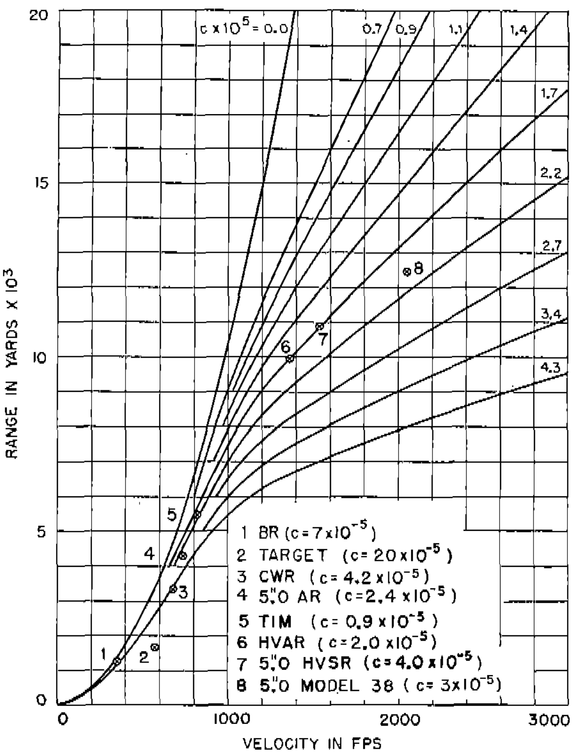

FIGURE 4. Maximum range as a function of velocity assuming Gâvre resistance function. Values of c apply to subsonic velocities.

TABLE 2. Ballistic quantities for spin-stabilized rockets.

$\delta \equiv$	angle of yaw; the angle between the axis of the rocket and the tangent to the trajectory.
$\delta_c \equiv$	equilibrium angle of yaw; yaw angle necessary if the spinner is to follow a smooth trajectory.
$\eta \equiv$	nozzle cant angle.
$\mu \equiv$	overturning moment coefficient; defined by equation (22).
$\nu \equiv$	"feet per turn"; distance traversed during one revolution.
$I_T \equiv$	total polar moment of inertia; equal to $(M + m_0)k^2$ before burning begins.
$K \equiv$	transverse radius of gyration; referred to an axis perpendicular to the rocket's axis of symmetry and passing through its center of mass; (ft). Total transverse moment of inertia of loaded round is $(M + m_0)K^2$.
$k \equiv$	polar radius of gyration; referred to the long axis of the rocket; (ft).
$R \equiv$	nozzle circle radius; perpendicular distance between the nozzle axis and the rocket axis; (ft).
$S \equiv$	stability factor; defined by equation (23); (dimensionless).
$s \equiv$	spin velocity; (radians per second).

using a rifled or spiral launcher or a rotating launcher, and allowing the blast to impinge on plates set at an angle either in or behind the nozzle, we shall use the term "spinner" to apply exclusively to finless rockets in which the rotation is imparted by ejecting the propellant gas through a number of identical nozzles, arranged in a circle and each inclined symmetrically to the axis of the rocket by a given angle. This device for imparting spin was the one most universally used by all the belligerents in World War II.

The velocity relations for spin-stabilized rockets have been worked out in reference 4, and the notation used in that report is summarized in Table 2. Remembering that V_g was defined as the effective velocity of the gas *relative to the nozzle*, it can be seen that, when the rocket is rotating so that the gas is ejected at an angle η with the axis of the rocket, only the component of the gas's momentum parallel to the axis is effective in pushing the rocket forward, so that the "effective gas velocity" is $V_g \cos \eta$, and the rocket gets slightly less forward momentum than if it were not rotating. The corrections to equations (4), (6), (8), and (13) consist obviously in replacing V_g by $V_g \cos \eta$.

A useful, but not quite accurate, expression for the angular velocity of the rocket can be derived by considering that the escaping gas exerts a thrust on each nozzle equal to its rate of change of momentum $V_g(dm/dt)$, resolving this force into its two components, and applying Newton's laws that force equals rate of change of linear momentum and torque equals rate of change of angular momentum. Then we have

linear momentum:

$$V_g \cos \eta \frac{dm}{dt} = (M + m)\frac{dV}{dt}; \tag{17}$$

angular momentum:

$$RV_g \sin \eta \frac{dm}{dt} = I_T \frac{ds}{dt}. \tag{18}$$

If it were possible to treat the combination of projectile and propellant as a rigid body and neglect the fact that the mass and radius of gyration of the propellant is constantly changing, we could substitute $(M + m)k$ for I_T and divide (18) by (17), obtaining

$$R \tan \eta = k^2 \frac{ds}{dt} \tag{19}$$

which integrates immediately into

$$s = \frac{RV \tan \eta}{k^2}. \tag{20}$$

[redacted]

This expression is strictly correct in the early part of burning provided that we use a value of the radius of gyration corresponding to the projectile plus propellant if the propellant rotates or corresponding to the projectile alone if the propellant does not rotate. It is not true, however, that the spin velocity continues to be proportional to the linear velocity; the spin increases more slowly than this, so that, later on in the burning, equation (20) always gives a value of s which is somewhat higher than the correct one. To derive a correct expression, one must know whether the propellant grain rotates at the same speed as the motor, at some slower speed, or not at all. Formulas applicable to these cases are discussed in reference 4. Experimental evidence is meager, but it appears that, for single-grain motors, the grain rotates almost as fast as the motor. The question is one of little practical importance, for the incorrect assumption that the angle of ejection of the gas is the same as the nozzle cant angle involves a considerably larger error. Despite the approximations involved in its derivation, equation (20) is useful for design purposes to give an estimate of the cant angle.

It is interesting to note that theory indicates the possibility of an equilibrium spin velocity [4,5] which cannot be exceeded by a rocket with a particular cant angle regardless of how high its forward velocity may become. The rocket could be made to spin so fast that the rotation would carry the nozzles sideways fast enough to allow the gas to flow straight back out of the nozzles and impart no further spin to the rocket. The equilibrium spin could be approached in practice only by a rocket with a very large nozzle-circle radius, and all rockets made to date fall far short of it.

To the approximation within which equation (20) is correct, the distance which the rocket travels while rotating once is a constant characteristic of the rocket. This quantity, designated "feet per turn," is given approximately by

$$\nu \equiv \text{"feet per turn"} = \frac{2\pi k^2}{R \tan \eta} \qquad (21)$$

if k and R are measured in feet. In practice, ν is always smaller than one would calculate from this formula because nozzles are so short that the effective nozzle cant angle (the angle which the ejected gas makes with the axis) is always somewhat smaller than η and cannot be measured. Hence ν, which is easily measured photographically, is taken as one of the fundamental ballistic constants.

21.4 FIN STABILIZATION

A long cylinder having its weight uniformly distributed along its length is in stable equilibrium flying through the air only when it is aligned perpendicular to its direction of motion, in which position its air drag is obviously very large. Since, in practically all rocket applications, we require that the projectile point in the direction of its motion so as to reduce air drag and land on its nose, it is necessary to stabilize it in this position. A cylinder flying through the air nose-on is in unstable equilibrium. If it acquires a slight yaw, that is, if the direction in which it is pointing and that in which it is moving begin to differ by a small angle, then the aerodynamic forces acting at each point of the surface cease to be uniformly distributed around the circumference. It is always possible, in such a case, to find a single force which, if applied at the proper point, will produce the same effect on the cylinder as the sum of all the complicated aerodynamic forces distributed over the surface. The point of application of this hypothetical force is called the center of pressure. It always happens that, unless the mass of the cylinder is concentrated very close to the nose, the center of pressure is forward of the center of mass so that the torque produced by the aerodynamic forces tends to increase the yaw (i.e., it is an "overturning moment") and cause the cylinder to tumble. To prevent this from occurring, two alternatives are available. Either we can arrange that the center of pressure be behind the center of mass so that the moment of the aerodynamic forces becomes a "righting moment," or we can spin the projectile so that the overturning moment combined with the gyroscopic effect causes the axis of the projectile to rotate around the direction of motion with a constant yaw instead of tumbling.

On a shell, the aerodynamic forces always produce an overturning moment. Some rockets, notably the 3.5-in. aircraft rocket with a solid steel head, have their centers of mass so far forward that the aerodynamic forces produce a righting moment even in the absence of fins, at least for large yaws after the propellant is consumed. In no case, however, is this righting moment large enough to produce the requisite stability without the necessity of having fins at the rear end of the rocket. The presence of fins increases the aerodynamic forces on the rear relative to those on the nose, and thus larger fins move the center of pressure farther back and increase the stability. Stability can be expressed quantitatively by the "eccentricity," defined as the ratio of the distance between the center of mass and the center of pressure to the length of the rocket, but it is more useful and customary to give the "yaw oscillation distance" σ. As its name indicates, σ is the distance the rocket travels while executing one complete oscillation from maximum yaw back to maximum yaw in a particular direction. It is discussed in greater detail in Chapter 24. We need merely note here that a small value of σ characterizes a stable rocket and it is desirable to have finners as stable as possible.

21.4.1 Dispersion of Finners

The most exasperating thing about fin-stabilized rockets is the infrequency with which they go in the direction that they are aimed. Their inaccuracy arises primarily from the failure of the line of thrust of the jet to pass through the center of mass of the rocket. This causes the rocket to rotate during burning about an axis through the center of mass perpendicular to the trajectory, with the result that the thrust of the motor is changed from its initial direction as determined by the orientation of the launcher. The perpendicular distance between the center of mass and the line of thrust (usually measured in thousandths of an inch) is called "malalignment," and a major portion of the effort in designing and manufacturing a finner is directed toward keeping it as small as possible.

The malalignment may vary in magnitude and direction during burning, but for theoretical analysis it is usually assumed to be a constant. In this case, it tends continually to increase the yaw of the rocket in a particular direction, and, since the yaw changes the direction of the thrust, a deflection in that direction results. What the direction of this yaw is depends upon the orientation of the rocket on the launcher, so that the directions are randomly oriented, left and right orientations being equally probable. Thus malalignment does not change the center of impact of a large number of rounds, but introduces a dispersion about this center which is roughly proportional to the malalignment. It would be expected, and was early demonstrated experimentally, that, after burning, a rocket continues in the direction it had at the end of burning, and no further inaccuracy is introduced.

The theory of dispersion is discussed at greater length in Chapter 24; its predictions are summarized as follows:[k]

1. For relatively low-velocity rockets having short burning times (e.g., the ASR and BR),[l] the burning distance is considerably less than half the yaw oscillation distance. The yaw caused by the malalignment, therefore, continues to increase all during the burning so that the deflection at the end of burning is approximately proportional to the burning time. Such rockets exhibit a marked decrease in dispersion with increasing temperature because of the shorter burning time. If a very accurate rocket of this type is desired, its burning time must be made short enough so that a large fraction of the burning takes place on the launcher.

2. For high-velocity rockets such as the forward-firing aircraft rockets, however, the fin size rather than the burning time is the most important factor in determining the dispersion. The reason for this is that the restoring torque due to the fins begins to become appreciable fairly early in burning and opposes the efforts of the malalignment to increase the yaw. The burning time is usually long enough so that the burning distance is somewhat longer than half the yaw oscillation distance, so that, before the malalignment torque ceases, the rocket has had time to reach a maximum yaw, return to zero yaw, and begin to yaw in a direction opposite to that induced by the malalignment. In the case of extremely long burning times, several oscillations may take place during burning. In either case, the final deflection is considerably less than that which would correspond to the maximum yaw of the rocket. Changes in burning time have only a minor effect on the dispersion.

3. For cases intermediate between 1 and 2, it is necessary to apply the theory in more detail (see Chapter 24).

The reason that the small malalignment torque

[k] Dispersion theory is treated in detail in reference 6 and is summarized in reference 7.

[l] 4.5-in. *barrage rockets* [BR] of more than six types existed, three of which were assigned Mark numbers. Since their basic design was similar we shall refer to them simply as the BR in cases where the differences are not involved. See Figure 3 of Chapter 18.

is able to rotate the rocket appreciably is, of course, that the stability of a finner depends upon its having a velocity relative to the air so that an aerodynamic torque exists tending to reduce the yaw. When starting from zero air velocity, the rockets are stabilized only by their launchers. If the rocket is headed into a high-velocity wind at the moment of firing, the fins are able to stabilize it from the beginning, and much lower dispersion results. This is the situation in forward firing from aircraft, and accounts for the facts that the dispersion of the same rocket air-fired is usually between 0.5 and 0.1 of its value when ground-fired and that aircraft rockets are designed with large fins so that their stability is large.

21.5 SPIN STABILIZATION

That a projectile can be stabilized by rotation even though the center of pressure is ahead of the center of mass is a consequence of the bizarre behavior of a gyroscope, which moves at right angles to the direction in which it is pushed. Stated more accurately, the rule is: if a gyroscope is rotating about a particular axis (vertical, say) and a torque is applied which tends to rotate it about an axis perpendicular to its spin axis (east), the result is a motion (precession) about the third mutually perpendicular axis (north). The directions of these axes are most conveniently remembered by imagining one's self standing behind the rocket (a very unsafe place to be except in imagination) and looking along its axis. Then, if the rocket is spinning to the right (clockwise), as is assumed throughout this discussion, a force tending to move the nose up results in motion of the nose to the right; a force tending to move the nose to the right results in motion down, etc. Thus, if a rocket has a yaw of, say, 1 degree, the overturning moment combined with the gyroscopic effect leaves the magnitude of the yaw unchanged but causes its direction to rotate clockwise around the trajectory.

The motion of a spinner is determined by the combination of the gyroscopic action with the various forces and torques which act upon it. Since there are four distinct types of forces and four of torques, the complexity of a spinner's motion is so great that even now their action throughout the trajectory is very incompletely understood. The fairly extensive theoretical work which had been done on the motion of shells was, of course, partially applicable to rockets, but the addition of the jet force during burning and the much greater relative length of rockets introduced new and complex phenomena which had not been observed with shells. During the last three years, much progress toward understanding them has been made, but they still present one of the most extensive and potentially fruitful fields for further research in rocketry. For the details of the theory, the original papers should be consulted. These are summarized a little more fully in Chapter 25, but in the following paragraphs we shall attempt to understand qualitatively the factors influencing a spinner's motion in order to see what points are important in design, ignoring, for the most part, the manifold complications.

21.5.1 Stability Factor and Rocket Design

If the rocket is moving through the air with a velocity V, it is subject to a torque (called the "aerodynamic overturning moment") tending to make it tumble. As long as the velocity is less than about 800 fps, the magnitude of the torque is given by

$$\text{Overturning moment} = \mu V^2 \sin \delta, \qquad (22)$$

where μ is the "overturning moment coefficient" and δ is the yaw angle. Whether this torque will be able to cause the rocket to tumble depends on the magnitude of the gyroscopic forces, which we can increase to any desired value by increasing the spin and by making the rocket relatively shorter and fatter. To express this fact precisely, we have a quantity called the "stability factor" which gives the ratio of the gyroscopic to the aerodynamic forces and is defined by the expression:

$$S = \frac{mk^4s^2}{4K^2\mu V^2}. \qquad (23)$$

Evidently we would expect that the rocket would be stable if $S > 1$, and this is actually the case for shells. Because of their greater length, rockets require a somewhat larger value, probably about 1.5.

A study of the expression for S reveals several important facts about spinner design. Suppose we wish to design a rocket of a particular diameter.

Then the "polar radius of gyration" k is fixed, since in practice it is almost impossible to change it much from a value of about 0.27 times the diameter. The spin velocity s is at our disposal, but a very definite upper limit on it is set by the centrifugal force which the motor tube and the propellant will stand. Now suppose that we have decided upon the length of our rocket as well as its caliber. Then the "transverse radius of gyration" K is also fixed, and we are left with the stability factor a function of s/V. As discussed in Section 21.3, this ratio, and hence the stability of the rocket, decreases steadily during the burning, so that, if the burnt velocity is too high, the rocket will become unstable sometime before the end of burning and begin to gyrate wildly, lose velocity quickly because of the enormously increased drag, and come to earth with a completely unpredictable orientation far from the original line of fire. If we are already spinning the rocket as fast as we dare, the only alternatives are to accept a lower velocity or to shorten the round so that S is increased by the decrease in K. Evidently, then, the higher the velocity required, the shorter the round will have to be. If we wish to exceed the velocity of sound, the problems are aggravated by the fact that μ, like the deceleration coefficient, ceases to be a constant in this vicinity and increases rapidly, so that still stubbier rockets will be required to keep S above the critical value. Apparently spinners for aircraft forward firing must have especially rapid spin and short length.

It would be interesting to know just what the maximum possible length of spinners is for various calibers, velocities, and shapes, but no specific investigation of the point was made at CIT because it was not of sufficient practical utility at the time. It is known that supersonic 5.0-in. rockets approximately 7 calibers long become unstable near sonic speed, but it may be that the instability could be cured by higher spin if it were attainable without grain fracture caused by centrifugal force. No difficulty was encountered in stabilizing 6-caliber 5.0-in. spinners. On the other hand, subsonic 3.5-in. spinners are adequately stable at 7 calibers length, although it is open to question whether they would be so at higher velocity.

An upper as well as a lower limit to the permissible stability factor exists in most cases. If the trajectory is very short and relatively straight and the rocket is not required to change its orientation, it may be desirable to have a very high stability. For example, the 5.0-in./14 GASR Model 39A,[m] designed for forward firing from aircraft, has a stability factor in ground firing of approximately 6, but rockets for barrage must have a much lower stability. If S is extremely large, a projectile fired at high angles will be so "stiff" that aerodynamic forces cannot turn it at all, and it will maintain its original orientation, landing base down.[n] At some lower value of S, the projectile will be able to turn rapidly enough to follow a relatively flat trajectory and land nose first; but, as the quadrant elevation is increased, it will have to turn more rapidly to follow the trajectory at its peak, and eventually a critical angle will be reached at which it is too "stiff" to follow over the peak, and instability results. To understand this effect, let us examine in greater detail the mechanism by which a spinner keeps oriented along its trajectory.

Consider a rocket which has been perfectly launched so that it is not wobbling or yawing. As soon as it leaves the launcher, gravity begins to act on it, causing the trajectory to become increasingly curved downward and effectively giving the rocket an up yaw. Since the aerodynamic moment is an overturning one, this yaw tends to lift the nose and it precesses to the right. With a yaw to the right, the precession moves the nose down, which is what is required to point it along the trajectory, and the rocket settles into a stable state in which it is yawed to the right at just the angle necessary to give sufficient overturning moment so that it precesses fast enough to follow the trajectory. In an actual case, of course, the mallaunching, the malalignment, and the dynamic unbalance will produce an initial yaw and a wobble so that the rocket will not have the equilibrium yaw. It will, however, oscillate about this yaw, which is a stable position, as can easily be seen by considering the moments introduced if the yaw deviates from it by a small amount.

If the rocket is spinning rapidly, it will be hard to turn, and the equilibrium yaw will have to be large

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[m] 5.0-in./14 GASR Model 39A is the CIT designation for a round which was developed in the summer of 1945 but did not receive a Navy designation. The "14" denotes its approximate velocity (almost 1,400 fps) and the letters stand for "General-purpose Aircraft Spin-stabilized Rocket." See reference 8 and Section 20.2.6 of the present volume for further details.

[n] This phenomenon has been observed with shells but not, so far as is known, with rockets, probably because sufficiently stable rockets have not been fired at high enough angles.

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in order to obtain sufficient torque to turn it. The theoretical expression for the yaw,

$$\delta_e = \frac{2\,K^2 g \cos\theta}{k^2 s V(1 - \sqrt{1 - 1/S})} \approx 4g \cos\theta \, \frac{K^2}{k^2} \, \frac{S}{Vs}$$

$$= mg \cos\theta \, \frac{k^2 s}{\mu V^3}, \tag{24}$$

shows also that it is largest where the velocity is least, i.e., at the peak of the trajectory, both because the curvature of the trajectory increases and because the aerodynamic forces are reduced. For high-angle fire, the peak yaw must be quite large. If the yaw exceeds a certain critical value (about 6 degrees is typical), other factors enter which make the rocket unstable, and it begins to gyrate wildly or, as we usually say onomatopoetically, to "wow-wow." The exact cause of this instability is complicated and not well understood; it is discussed in somewhat greater length in Chapter 25.

That a spinner goes through life with its nose pointed to the right causes it to be deflected from its "normal" trajectory in two ways. The *lift* gives it a drift to the right, and the *Magnus force* pushes upward so that the range is increased beyond what would be expected from the velocity and the drag. (See Chapter 25 for definitions of these forces.)

21.5.2 Spinner Trajectory during Burning

The motion during burning is much more difficult to visualize than that after burning because of the addition of the jet force. Two factors—wind effect and mallaunching—are important during this period. The theory of wind effect has been calculated,[9,10] and its complexity can be seen from the following summary.

Wind		Deflection Produced
Cross wind from left:	Light	Right and down
	Medium	Left and down
	Strong	Left and up
Down-range wind:	Light	Left and down
	Medium	Left and up
	Strong	Right and up

The definition of a "strong" wind depends upon the round, its lower limit ranging from about 25 to 40 fps. The wind effect is complicated principally because it is nonlinear, and its magnitude is usually in the range of one to several mils per foot per second so that it is too large to be ignored. Because the burning time is so short, gustiness in the wind can produce large differences in effect between rounds, thus introducing dispersion. Their sensitivity to wind is thus a severe limitation on the accuracy of spinners.

The mallaunching effect is complicated for quite a different reason. Mallaunching is the name given to any angular acceleration about a transverse axis which the round acquires as a result of interactions with the launcher. Tip-off is the most frequently encountered example, tending to give the round an angular velocity around a horizontal axis. As the nose drops, the precession moves it to the left, and the resulting left yaw makes it continue precessing upward. Thus the nose moves in a spiral relative to the center of mass, and the rocket finishes burning in a position below and to the left of its theoretical position for no tip-off. Since the drift after burning is to the right, low-angle rounds land to the left of the range line, and high-angle rounds land to the right.

If the launcher constrains the rocket from moving in any lateral direction, as is usually the case with spinners, the mallaunching need not be downward. Either malalignment or dynamic unbalance may give the round a transverse rotational velocity in some other direction. Theory indicates that, for a given degree of mallaunching, the resulting dispersion is reduced by a longer launcher (i.e., by a faster spin at launching). However, with some launchers it occurs that the malalignment is approximately proportional to the spin velocity at launching because of the larger forces involved, so that increased launcher length does not result in greater accuracy. For this reason, it is very difficult to determine what type of launcher is best, and many diverse types and lengths have been tried. Usually the choice has fallen on a simple, relatively short launcher for obvious tactical reasons because longer and more complicated ones have not given appreciably smaller dispersions.

21.5.3 Special Purpose Spinners

A further result of the complexity of spinner trajectories and the large number of factors influencing them is that a spinner should be tailor-made to a particular purpose in order to function with best effect. A "Jack-of-all-trades" spinner would probably indeed be "master of none." In the

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early days of spinner development, much effort was expended in an attempt to work out a compromise round which could be used both for accurate fire with a flat trajectory and for barrage at relatively high angles. The rounds developed for barrage had a stability factor of about 2 and were able to follow over a trajectory as high as about 55 degrees to 60 degrees. Although satisfactory for barrage, they could not be made more accurate than about 8 mils, principally because cross winds produced deflections of about 2 mils per fps, so large an effect as to make the fire control problem virtually insurmountable in cases where accurate fire is desired. Since the cross-wind effect is approximately inversely proportional to the stability factor, which, in turn, depends on the square of the spin velocity, it appears to be desirable to use fast-spin rockets with flatter trajectories for most applications requiring greater accuracy.[11] On the other hand, greater attention must be paid to mallaunching and dynamic balance for such fast-spin rounds, so that the increased accuracy does not come cheaply.

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Chapter 22

DESIGN OF ROCKET PROPELLANT CHARGES

By *C. W. Snyder*

22.1 GENERAL REQUIREMENTS FOR A ROCKET CHARGE

THE GENERAL PROBLEM of designing a propellant charge is a very large one, involving choice of the propellant composition to be used, a grain shape, the number of grains in the charge,[a] the method of extruding or otherwise forming the grain, etc., as well as the specific problems of fitting the charge to the motor contemplated. The general problems are beyond the scope of this part of the report. We shall take the viewpoint of the man who wishes to design a rocket with particular performance characteristics using a propellant of fixed composition—specifically ballistite, since this is the only propellant which has been used in quantity in this country—although the discussion will be generally applicable to other propellants as well. We shall not be concerned with the general theoretical treatment of propellant performance, more complete treatments of which will be found in *The Interior Ballistics of Rockets*,[1] and in *Rocket Fundamentals*.[2]

The charge designer is usually called upon to meet four specifications: caliber, impulse, temperature range, and burning time. The caliber of the motor fixes the grain's external diameter, which must be just slightly smaller than the tube's internal diameter in the case of single-grain charges or, for multiple-grain charges, must be such that the grains nest properly in the tube. With a fast-burning propellant such as ballistite, single-grain charges are preferable, and multiple-grain charges are used only when either (1) it is not feasible to make a grain as large as the motor, or (2) the shorter burning time obtainable with several thin-walled grains is desirable. Both these considerations entered in the cases of the Army's 4.5-in. rockets using solvent-extruded powder (which cannot be made in web thicknesses exceeding 0.4 in.) and of the Tiny Tim (which was too big for the available propellant extrusion presses). Unless otherwise noted, we shall consider only single-grain motors.

[a] The term "propellant charge" is applied to the powder which furnished the energy for propulsion in its final state ready for assembly into the rocket motor. The term "propellant grain" signifies a single extruded or molded piece of propellant of whatever size (it may even be several inches in diameter and several feet long) either finished or unfinished. Thus a charge may consist of several grains, or of a single grain assembled with inhibitor strips or end washers, or simply of a single grain with nothing attached. In this chapter, we shall extend the customary meaning of the word "charge" to include not only the powder grain or grains, but also the igniter, grid, and other pieces intimately associated with the grain but not necessarily attached to it.

22.2 IMPULSE AND GAS VELOCITY

The impulse or integrated thrust delivered by the motor is given by equation (8) of Chapter 21 as the product of the mass of the grain by the effective gas velocity V_g. The theoretically attainable gas velocity depends on the propellant composition,[2a] and in practice its value for ballistite is approximately 7,000 fps, which is higher than that for most other propellants because of the larger heat of combustion of ballistite. For any particular motor, V_g must ultimately be determined experimentally by measurements of that rocket's velocity in the field, but in preliminary design calculations one may use a value determined from tests of a similar motor.

Gas velocities for typical CIT rockets are shown in Figure 1. It will be noted that V_g increases with temperature over most of the temperature range, the reason being partly that, when the propellant and metal parts are cold, more of the heat energy liberated is consumed in heating them up and less is available as kinetic energy of the gas to impart momentum to the rocket. That the curve usually turns down again at high temperatures results from the decreased strength of the propellant, more and more of the powder being broken up and expelled without being burnt. The decline at high temperatures is absent when the grain is not subject to large forces (from acceleration or pressure drop) and when high-strength propellant is used, as can be seen from the curves.

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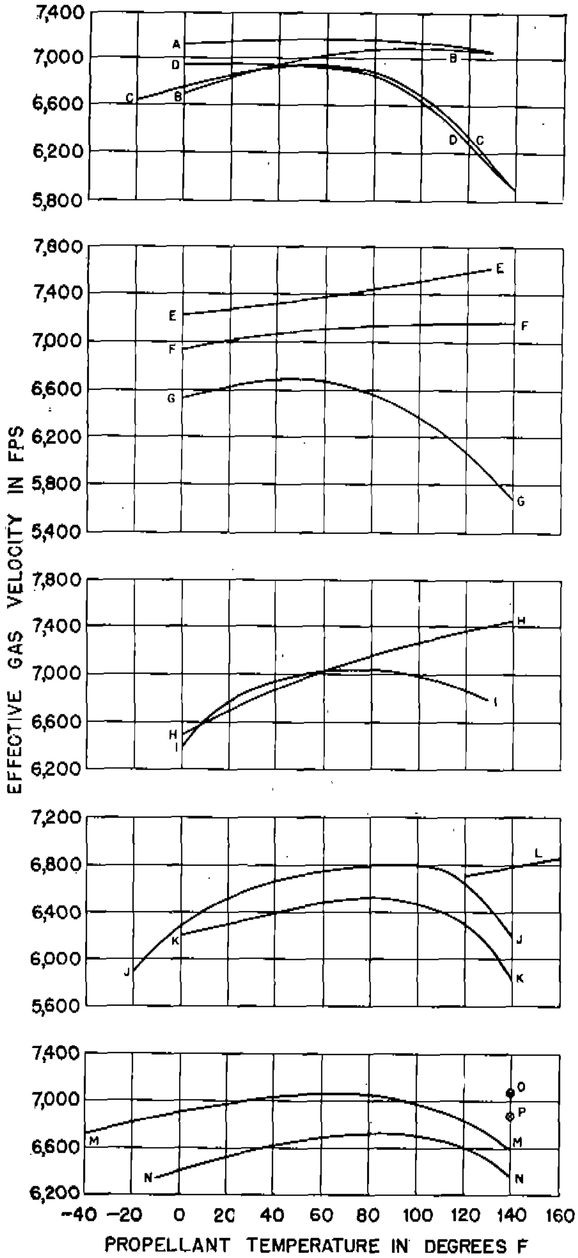

FIGURE 1. Gas velocities for various CIT rockets.

Effect of web thickness and acceleration, curves A, B, C, D:

Constant: Motor inside diameter 2.0 in.
Grain outside diameter 1.7 in. 3-ridge tubular
Grain weight 1.43 or 1.44 lb

Variable: Grain inside diameter, web thickness, and length
Projectile weight and acceleration

Curve A: 7.2-in. Rocket Mk 1 Mod 0 (antisubmarine rocket); 60.1 lb
2.25-in. Motor Mk 3 Mod 1
Mk 1 Mod 0 Grain, 1.70 × 0.59 × 11.6 in.

Curve B: 4.5-in. Rocket Mk 1 Mod 0 (barrage rocket); 28.7 lb
2.25-in. Motor Mk 7, 8, or 9 Mod 0
Mk 1 Mod 0 Grain, 1.70 × 0.59 × 11.6 in.

Curve C: 7.2-in. Rocket CIT Model 18 (demolition rocket); 61.8 lb
2.25-in. Motor CIT Model 14
Grain 1.67 × 0.90 × 14.9 in.

Curve D: 4.5 BR (special with short burning time); 29.5 lb
2.25-in. Motor CIT Model 7
Grain 1.7 × 0.9 × 14.5 in.

Remarks: Differences at low temperatures not understood. Note absence of acceleration effect at these low accelerations and marked drop in V_g for thin-web grains at high temperature.

Effect of length or internal K for thin-web grains, curves E, F, G:

Constant: Rocket type 7.2-in. retro rocket
Motor type 3.0-in. inside diameter with box grid
Grain shape 2.5 × 1.4 in. 3-ridge tubular
Projectile weight and acceleration (approximately)

Variable: Grain length and weight and internal K

Curve E: 7.2-in. Rocket Mk 7 Mod 0; 64.3 lb
3.25-in. Motor Mk 1 Mod 0
Mk 6 Mod 1 Grain, 8.7 in. long, 1.8 lb, $K_I = 37$

Curve F: 7.2-in. Rocket Mk 10 Mod 0; 67.3 lb
3.25-in. Motor Mk 2 Mod 0
Mk 7 Mod 1 Grain, 13.0 in. long, 2.8 lb, $K_I = 55$

Curve G: 7.2-in. Rocket Mk 12 Mod 0; 70.6 lb
3.25-in. Motor Mk 3 Mod 0
Mk 8 Mod 1 Grain, 19.6 in. long, 4.1 lb, $K_I = 82$

Remarks: Note efficiency at high temperature—extremely high for shortest grain, extremely low for longest grain, the latter due to thin web.

Effect of length or internal K for thick-web grains, curves H, I:

Constant: Rocket type 2.25-in. subcaliber aircraft rocket
Motor type 2.0-in. inside diameter with box grid
Projectile weight approximately 12 lb
Grain shape approximately 1.7 × 0.27 in. 3-ridge tubular

Variable: Grain length and weight and internal K

Curve H: 2.25-in. Rocket Mk 2 Mod 0
2.25-in. Motor Mk 12 Mod 0, $K_N = 225$
Mk 17 Mod 0 Grain, 8.5 in. long, 1.12 lb, $K_I = 77$

Curve I: 2.25-in. Rocket Mk 1 Mod 0
2.25-in. Motor Mk 10 Mod 0, $K_N = 190$
Mk 16 Mod 0 Grain, 12.5 in. long, 1.75 lb, $K_I = 121$

Remarks: Note better high-temperature performance shown in curve I than in curve G because of thicker web. Curve I represents the heaviest grain that has been successfully used in a 2.0-in. motor.

Effect of propellant strength and nozzle coefficient, curves J, K, L:

Constant: Rocket type 3.5-in. Rocket Mk 1 Mod 0 (AR)
Grain Mk 13 Mod 0, 2.74-in. cruciform, $K_N = 167$

Variable: Motor inside diameter at nozzle end of grain, internal K
Expansion ratio of the nozzle
Propellant composition and strength

Curve J: 3.25-in. Motor Mk 7 Mod 0, $K_I = 112$
Propellant JP; ultimate strength 270 psi at 140 F
Expansion ratio 4.0; nozzle coefficient 1.47

Curve K: 3.25-in. Motor Mk 6 Mod 0; $K_I = 130$
Propellant JP; ultimate strength 270 psi at 140 F
Expansion ratio 2.35; nozzle coefficient 1.40

Curve L: Same as curve J except:
Propellant RDS 1154.2; ultimate strength 700 psi at 140 F

Remarks: Note improved high-temperature performance with high-strength propellant. Effect of K_I is not apparent in these curves but shows up in burst frequency at high temperature.

Gas velocity of large motors, curves M and N, points O and P:

Constant: Grain shape 4.2-in. cruciform

Variable: Motor design and grain size
Propellant composition and strength

Curve M: 5.0-in. Rocket Mk 4 Mod 0 (HVAR)
5.0-in. Motor Mk 1 Mod 0
Mk 18 Mod 0 Grain, 24.0 lb, 39 in. long
Propellant JPN; ultimate strength 230 psi at 140 F

Curve N: 11.75-in. Rocket Mk 3 Mod 0 (Tim)
11.75-in. Motor Mk 1 Mod 0
4 Mk 19 Mod 0 grains; 36.3 lb, 60 in. long
Propellant JPN; ultimate strength 230 psi at 140 F

Point O: Same as curve M except:
Propellant JPH; ultimate strength 550 psi at 140 F

Point P: Same as curve N except:
Propellant JPH; ultimate strength 550 psi at 140 F

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From the viewpoint of external ballistics, V_0 is related to the velocity of the rocket [equation (6) of Chapter 21]; from the viewpoint of internal ballistics it is related to the pressure-time curve, which can be determined by static firing. Equation (13) of Chapter 21 shows that it is determined by the area under the pressure-time curve and has no relation to its shape.

22.3 PRESSURE-TIME CURVES

To determine the exact shape of the pressure-time curve required, we turn to the temperature-range specification, which usually reads "as large as possible," although for certain applications either the upper or the lower temperature limit may be more critical. With ballistite (and all other propellants in common use), the pressure in the motor during burning is considerably increased as the initial propellant temperature is increased, yet the maximum pressure developed at any point in the burning must never exceed the safe working pressure of the motor assembly at the highest propellant temperature to be encountered in service. On the other hand, in order to maintain satisfactory burning of the powder, the reaction pressure at the lowest firing temperature must not fall below a certain limit, normally about 300 psi. Thus it can be seen that the maximum temperature range over which the motor will function satisfactorily is obtained when the maximum reaction pressure at any given temperature is as low as possible and the minimum pressure at the same temperature is as high as possible. This condition is obtained when the pressure remains essentially constant throughout the reaction period, or, as we say, when the burning is "neutral.'

The situation is modified slightly by the fact that in many cases the motor walls are heated sufficiently by the propellant gas to decrease in strength toward the end of burning, and the maximum pressure which can be permitted is consequently lower at the end of burning than during the first part. But it has also been found that the minimum pressure at which satisfactory continuous burning can be maintained decreases as the motor walls are heated and is therefore somewhat lower during the latter part of burning than the pressure necessary to ignite the grain. These two considerations indicate that the most desirable pressure-time curve will usually be one which is slightly "regressive," that is, one in which the pressure decreases somewhat from the start to the end of the reaction. The exact amount of regression to produce optimum results in a given application must usually be determined by experiment, but it is seldom large.

22.4 EQUILIBRIUM PRESSURE IN A MOTOR

In order to see how the shape of a pressure-time curve can be controlled, we shall examine the factors which determine the motor pressure. The first of these is the burning characteristics of the powder. If any substance which does not require an exterior supply of oxygen to support its combustion is ignited simultaneously over its whole surface (and if no complications arise, such as obstructions to the free flow of gas away from certain portions of the surface), burning will proceed at the same rate at every point, so that each surface remains always parallel to its original position. The accuracy with which the burning takes place perpendicular to the surface is strikingly shown by partial burnings of grains, some of which are shown in Figures 9 and 11. If a small indentation (a serial number, for instance) is made in the surface, it will still appear—and be legible—although the surface may have receded $\frac{1}{4}$ in. or more from its original position. In this

TABLE 1. Internal ballistics quantities.

$\beta \equiv$	proportionality constant relating motor pressure to burning rate; it is a function of temperature.
$\rho \equiv$	propellant density.
$A_N \equiv$	nozzle throat area.
$A_S \equiv$	surface area of the grain.
$B \equiv$	linear burning rate of propellant.
$C_D \equiv$	nozzle discharge coefficient.
$D \equiv$	internal diameter of motor tube.
$d \equiv$	external diameter of grain (excluding ridges or inhibitors).
$\delta \equiv$	internal diameter of grain (cylindrical).
$K_I \equiv$	"internal K"; ratio of charge area to port area.
$K_N \equiv$	"nozzle K"; ratio of charge area to nozzle throat area.
$L \equiv$	grain length. $\lambda = 100\, L/K_I D$.
$M_P \equiv$	propellant grain mass.
$\dot{m} \equiv$	mass rate of discharge of gas through the nozzle.
$\dot{m}' \equiv$	mass rate of production of gas by the grain.
$P \equiv$	pressure at any point in the motor.
$P_N \equiv$	"nozzle pressure"; measured just to the rear of the nozzle end of the grain.
$t_b \equiv$	burning time.
$v \equiv$	volume of grain.
$W \equiv$	web thickness of grain.
$w \equiv$	web thickness. $w = W/D$.

type of burning, it is obvious that the amount of powder which disappears, and hence the amount of gas generated, will depend only on two factors: (1) the speed with which each surface recedes, i.e., the linear burning rate B, and (2) the instantaneous ignited surface area A_S of the grain.

Now the linear burning rate B is a function, unfortunately, of a considerable number of variables, including pressure, the initial temperature of the grain, the velocity of gas past its surface, and the radiation density in its vicinity. It is most critically dependent upon the pressure, and hence is frequently expressed by the approximate relation:[1a]

$$B = \beta\left(\frac{P}{1{,}000}\right)^n \tag{1}$$

where β is a constant for any particular temperature. For ballistite, the exponent n is approximately 0.7, and the variation of burning rate with pressure and temperature is as shown in Figure 2.

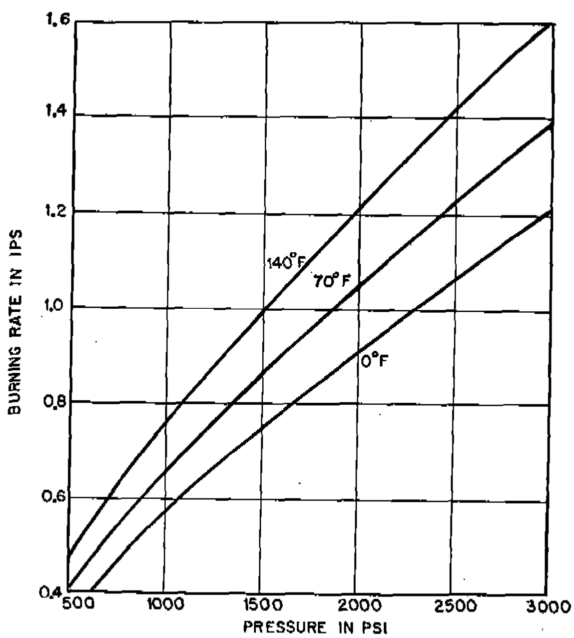

FIGURE 2. Burning rate of JPN ballistite.

As soon as gas begins to be generated by the burning of the charge, a pressure differential between inside and outside the motor appears, and gas begins to flow out through the nozzle at a rate which is proportional to the pressure difference and to the nozzle area. That is,

$$\dot{m} = C_D A_N P_N, \tag{2}$$

where A_N is the nozzle throat area, P_N is the excess pressure just inside the nozzle, and $\dot{m}$ is the mass of gas discharged per second. Except for very low pressures, such as are not usually used in rocketry, the proportionality factor C_D, called the "nozzle discharge coefficient," is constant for a given powder composition. If A_N is expressed in square inches, P_N in pounds per square inch, and $\dot{m}$ in pounds per second, its value for ballistite is[b]

$$C_D \approx 0.00065 \frac{\text{lb (mass)}}{\text{lb (force)} \times \text{sec}} \text{ or sec}^{-1}.$$

We have already seen that the rate of generation of gas by the charge depends on the linear burning rate B and the charge area A_S, so we can write

$$\dot{m}' = \rho A_S B, \tag{3}$$

putting the density of the propellant ρ in pounds per square inch. Since the burning rate increases with pressure, the pressure will rise rapidly when the motor is fired, until a value is reached such that the rate of gas flow through the nozzle is equal to the rate of gas generation, or

$$\dot{m} = \dot{m}' = C_D A_N P_N = \rho A_S B. \tag{4}$$

This relation enables us to solve for the equilibrium pressure.[c]

$$P_N \text{ (equilibrium)} = \frac{\rho B}{C_D} \cdot \frac{A_S}{A_N}. \tag{5}$$

Although the above treatment is only approximately correct, since we have neglected the pressure gradient inside the motor which causes the gas to flow toward the nozzle[d] (the pressure difference between the two ends may amount to several hundred pounds per square inch), it does correctly show that, except for constants depending only on the propellant composition, the instantaneous equilibrium pressure is determined by the ratio of the ignited area of the charge to the area of the nozzle throat. This ratio is the most important design constant in the internal ballistics of rockets and is denoted by

[b] Theoretical derivations of the value of C_D are given in *The Interior Ballistics of Rockets*,[1b] and in *Rocket Fundamentals*.[2a] The nozzle discharge coefficient C_D must be distinguished carefully from the aerodynamic drag coefficient in Chapter 21 which was denoted by the same symbol. Normally both quantities will not appear in the same report.

[c] The product ρB, in units of lb/ft²-sec or slugs/ft²-sec, is called the burning rate in much of the literature of internal ballistics.

[d] A more nearly exact relation is given in *The Interior Ballistics of Rockets*.[1c]

[redacted]

the symbol K_N and called "the nozzle K" or simply "the K" of the motor.

22.4.1 Advantages of Tubular Grains

This analysis shows that, unless the burning rate varies during the reaction, a grain will be neutral-burning if it is "geometrically neutral," that is, if its surface area remains constant during the reaction. The burning rate is not strictly constant, but pressure-time curves do follow area-time curves fairly closely as shown in Figure 3. A solid cylinder burns regressively, a tube ignited only on its inside surface is progressive-burning, and a tubular grain which burns inside and outside but not on the ends is, geometrically, neutral because the internal radius and area increase at the same rate as the external radius and area decrease. It is primarily because of this characteristic of neutral burning that tubular grains are used in most rockets. Evidently, if we wish to introduce a little regression into the burning of a tubular grain, we can allow one or both of the ends to burn so that the length decreases during the reaction. If we do not wish the ends to burn, we must prevent the propellant gas from touching them by cementing on a plastic "inhibitor" disk.

Tubular grains are popular for two other reasons also. First, they are easy to extrude and require relatively little processing after extrusion; hence they are inexpensive. Second, they burn up completely because their wall thickness is the same at every point. As the walls become thinner uniformly during burning, their thickness becomes zero everywhere simultaneously. The smallest distance between two adjacent burning surfaces is called the "web thickness," and most shapes of grains have some portions that are thicker than the "web." After the web is burned through, the charge area is so small that it cannot maintain the pressure required for continuous burning, and the pieces remaining, called "slivers," either are ejected unburnt or smolder slowly and contribute nothing to the momentum of the rocket. Obviously the burning time of a grain, which was the last specification the grain designer had to meet, is given by

$$t_b = \frac{w}{2B}, \tag{6}$$

where w is the web thickness. The factor 2, which appears because burning takes place from both sides of the web, is omitted for grains which burn only on the interior surface.

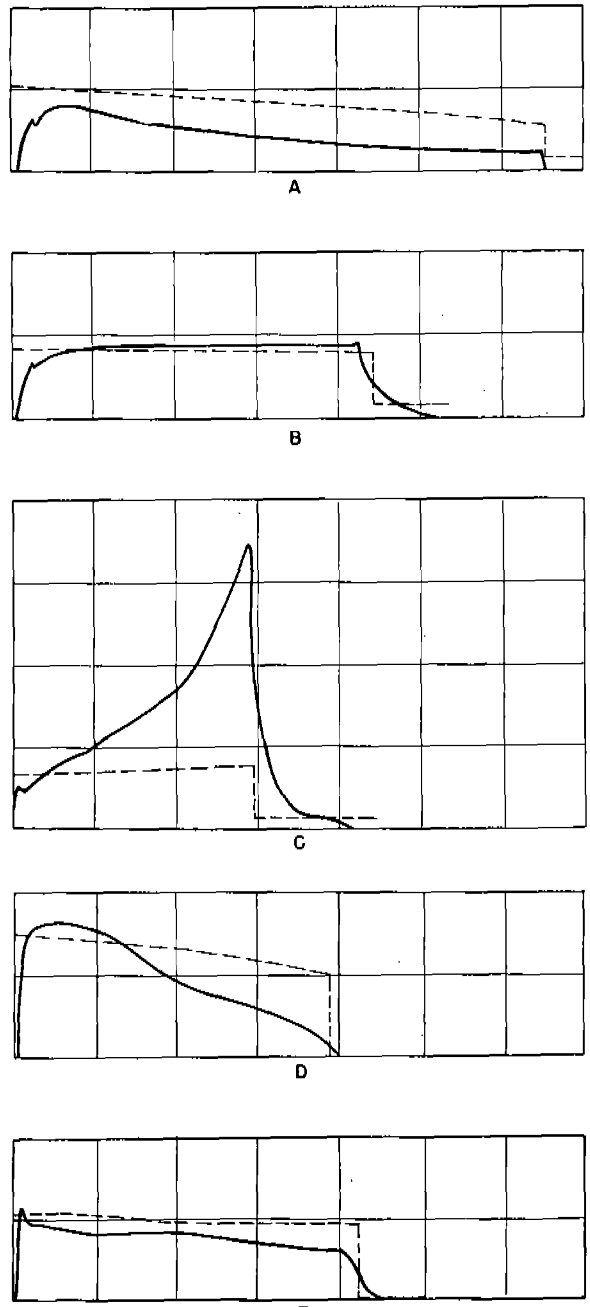

FIGURE 3. Comparison of pressure-time curves (solid lines) and area-time curves (dashed lines) for: (A) cruciform grain with no arms inhibited, (B) cruciform grain with two arms inhibited full length, (C) cruciform grain with all four arms inhibited full length, (D) thick-web tubular 3-ridge grain with radial holes, (E) thick-web tubular 3-ridge grain with rod stabilization and no radial holes. All grains are inhibited on both ends.

[redacted]

22.4.2 Pressure Drop

Returning to the question of what determines the pressure in a rocket motor, we can apply again the logic by which we found that the nozzle pressure is fixed by the constant K_N. In the conventional type of rocket, all the gas is discharged at the rear, and, if any particular mass of gas is to be urged toward the nozzle as it leaves the charge surface, a pressure gradient must exist along the length of the charge. The pressure at any point must depend on the rate at which gas is generated ahead of it (proportional to its distance from the front end of the charge) and the port area through which it can flow (which is usually the same for all points along the charge). Hence the difference in pressure between the front and rear ends of the grain depends on the total area of the grain and the port area between the grain and the motor tube. This space at the nozzle end of the grain acts like a secondary nozzle, and for it we define a "K" which is

$$\text{"Internal } K\text{"} = K_I = \frac{\text{charge area}}{\text{port area}}. \qquad (7)$$

As long as K_I is much less than K_N, it is of little importance, but, if it begins to be too large, then the pressure drop along the grain also becomes large, and troubles appear at high temperatures.

22.4.3 Temperature Sensitivity

Increasing the temperature of the propellant causes it to burn faster because the constant β in equation (1) is a function of temperature, increasing by about one-third between 0 F and 140 F.[e] If the burning rate increases, so also does the equilibrium pressure in the same proportion according to equation (5). But this increased pressure causes a still further increase in burning rate, which in turn increases the pressure again. Thus, for a constant nozzle size, the variation in equilibrium pressure with temperature is relatively large, pressures at 140 F being typically from 2.5 to 3.5 times those at 0 F with ballistite. This large temperature sensitivity is one of the primary disadvantages of ballistite as a rocket fuel. Development of less temperature-sensitive propellants will simplify the problems of rocket design greatly and make possible lighter motors and increased temperature range.

[e] Values of β for 0 F, 70 F, and 140 F for various propellants are tabulated in *The Interior Ballistics of Rockets*.[1d] The change with temperature varies from 15 per cent for 218B composite propellant to 48 per cent for JP, the original ballistite composition used in CIT rockets.

22.5 CALCULATION OF MOTOR PERFORMANCE

We have seen that the pressure in a motor depends in a complicated fashion on the initial temperature of the propellant (which determines the burning rate at a given pressure), the nozzle K (which sets the equilibrium pressure for a given burning rate in the absence of a pressure drop along the grain), and the internal K (which causes a pressure drop along the grain and thus alters the nozzle K required). Because of the interrelation of these factors, a calculation of the equilibrium pressure in a motor can only be made by successive approximations. The relation of the quantities to the basic thermodynamic properties of the propellant has been extensively investigated by the propellant section of the project, and formulas have been derived for calculating the pressure distribution in a motor by means of certain simplifying assumptions.[1c,3] Calculations based on them must usually be altered by experimental correction factors to take account of such complications as heat turbulence and heat loss to the surroundings, which are too difficult to treat analytically. Practical calculations of motor pressures, then, are made with semiempirical relations based on the theory. For convenience, it is desirable to have the relations in graphical form. For ballistite, such graphs have been published in reference 7, and their use is illustrated in the following sample calculation. Of necessity, they represent rather ideal conditions, but for the calculation of maximum pressure they have been found to be fairly reliable.[4]

As an example of the calculations involved in designing a grain, let us suppose that we are required to meet the following specifications:

1. Motor tube 4.625-in. inside diameter;
2. Velocity 1,000 fps with 100 lb of metal parts;
3. Maximum front pressure 3,000 psi at 130 F;
4. Burning time 0.70 second at 70 F.

We shall for simplicity use a cylindrical grain inhibited on the ends, even though, as discussed in Section 22.6, such a simple shape is seldom used. The alterations in the method of calculation for other grain shapes are relatively obvious. For the various ballistic constants, we shall take the values determined for JPN ballistite.

Calculation of Powder Weight. Assuming a gas velocity of 7,000, we have from equation (6) of Chapter 21

$$m_0 = \frac{MV_0}{V_g - \frac{1}{2}V_0} = \frac{100 \times 1{,}000}{7{,}000 - 500} = 13.316. \quad (8)$$

Calculation of Burning Time. The burning time will depend on the pressure at which we decide to operate the motor and the web thickness. Since neither of these has been determined, we have insufficient data to find the burning time. If we look at the curves of pressure vs nozzle K and internal K in the range of K's which we expect to be using, we will find that pressures at 70 F are slightly less than half those at 130 F. Hence, provisionally, we take a pressure of 1,375 psi. The burning rate is, then,

$$B = \beta\left(\frac{P}{1{,}000}\right)^n = 0.651 \times 1.375^{0.7} = 0.814 \text{ ips} \quad (9)$$

so that the web thickness is

$$w = 2Bt_b = 2 \times 0.70 \times 0.814 = 1.14 \text{ in.} \quad (10)$$

Calculation of Grain Shape. The mass of a cylindrical grain is given by

$$m_0 = \frac{\pi}{4}(d^2 - \delta^2)L\rho, \quad (11)$$

where ρ is the density (for ballistite, approximately 100 lb per cu ft or 0.058 lb per cu in.), L is the length, and δ and d are the inner and outer diameters respectively. In terms of the web thickness w, this is

$$m_0 = \pi\rho Lw(d - w) \quad (12)$$

or

$$L(d - w) = \frac{m_0}{\pi\rho w} \quad (13)$$

$$L(d - 1.14) = \frac{13.3}{1.14\pi \times 0.058} = 64.3. \quad (14)$$

Choosing either the outer diameter or the length, we can calculate the other from this equation. Since we would like to make the motor short in order to save weight, the diameter should be made as large as possible consistent with good performance. This is limited by the necessity for leaving adequate port area for the gas which is generated on the outside surface, but the maximum permissible diameter would have to be determined by experiment. The factors involved are plotted in Figure 4, including the grain length, the internal K for the motor as a

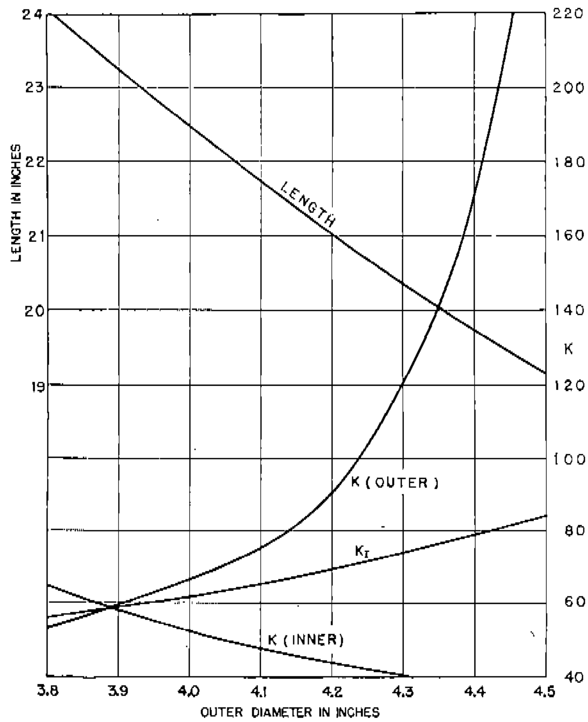

FIGURE 4. Relation of length to internal K for typical tubular grain with no radial holes.

whole and the internal K's for the central perforation and the outer annular channel separately.

The internal K's are calculated as follows:

Inner channel:

$$A_S = \pi\delta L = \text{grain area};$$

$$A_P = \frac{\pi}{4}\delta^2 = \text{port area};$$

$$K \text{ (inner)} = \frac{4L}{\delta}. \quad (15)$$

Outer channel:

$$A_s = \pi dL;$$

$$A_p = \frac{\pi}{4}(D^2 - d^2);$$

where D is the internal diameter of the motor tube.

$$K \text{ (outer)} = \frac{4dL}{D^2 - d^2} = \frac{4dL}{21.4 - d^2}. \tag{16}$$

Motor as a whole:

$$K_I = \frac{4L(d + \delta)}{21.4 - (d^2 - \delta^2)}. \tag{17}$$

The ballistically ideal grain might be thought to be the one in which the two internal K's were equal; from the graph it would have an external diameter of 3.89 in. and a length of 23.3 in. A saving of 3 in.

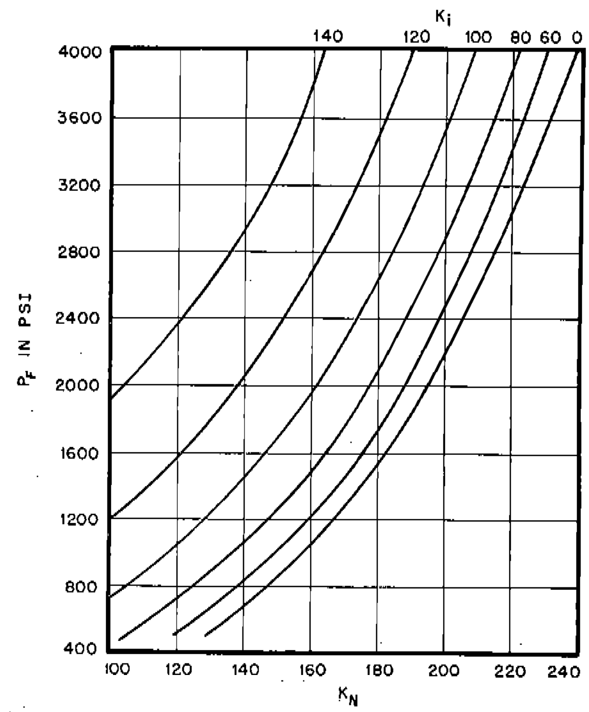

FIGURE 5. Graph of front pressure vs nozzle K for 130 F. Time = 0.02 second.

or more in length can be made, however, if the resulting high value of the internal K for the annular channel does not cause trouble, as in fact it does not.[5] The calculation of K (outer) assumes that all the gas moves toward the rear of the motor. Actually, of course, if the K (outer) is much higher than K (inner), the gas generated near the front of the outer surface will move forward and escape through the central perforation, thus effectively lowering the K (outer) and increasing the K (inner). K_I is, therefore, the important design parameter. The performance of grains of a given weight with various lengths, diameters, and web thicknesses are summarized in *The Interior Ballistics of Rockets*,[11] reference 6.

For the purpose of the present example, let us assume that we have decided on a grain 19.6 in. long, since it is clear from the graph that this is close to the shortest possible grain that will work. This choice determines the dimensions of the grain to be 4.42 in. by 2.14 in. and gives internal K of 80, which is convenient for calculation.

It remains to calculate the nozzle throat diameter required to give a front pressure of 3,000 psi at 130 F. From the graph in Figure 5,[f] a nozzle K of 202 corresponds to this pressure when $K_I = 80$.

Since the grain's burning area is

$$A_S = \pi L(d + \delta) = 19.6\pi(4.42 + 2.14) = 404 \text{ sq in.}, \tag{18}$$

the nozzle throat area is

$$A_N = \frac{A_S}{K_N} = \frac{404}{202} = 2.0 \text{ sq in.} \tag{19}$$

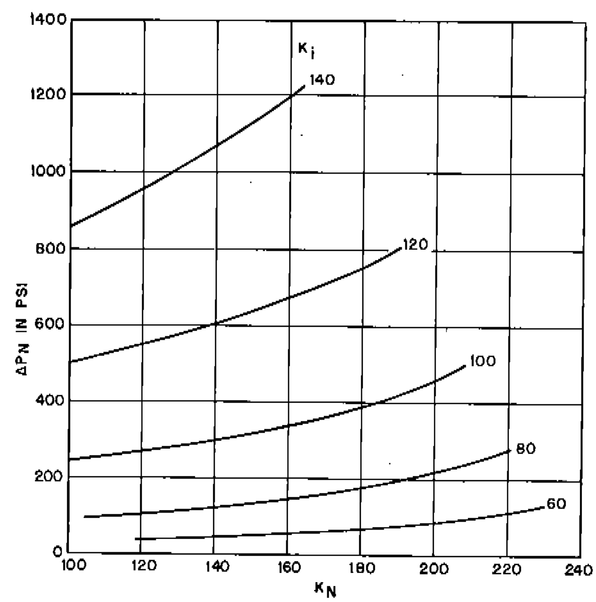

FIGURE 6. Graph of pressure difference between front and nozzle ends vs nozzle K for 130 F. Time = 0.02 second.

This requires a single nozzle 1.6 in. in diameter or multiple nozzles of correspondingly smaller size.

Further information about the grain's performance can be gotten from the graphs of Figures 6 and 7 and

[f] Figure 5 is taken from reference 7, which gives similar curves for front pressure, nozzle pressure, and pressure difference as functions of K_N and of temperature.

[redacted]

also from reference 7. For example, the pressure drop along the grain at 130 F is 225 psi, giving a total force on the grain

$$F = \Delta P \times A_{\text{end}} = \Delta P \times \frac{\pi}{4}(d^2 - \delta^2)$$
$$= 2{,}640 \text{ lb.} \tag{20}$$

Front end pressures will be slightly higher than those read from Figure 7 for $K_N = 200$ and $K_I =$

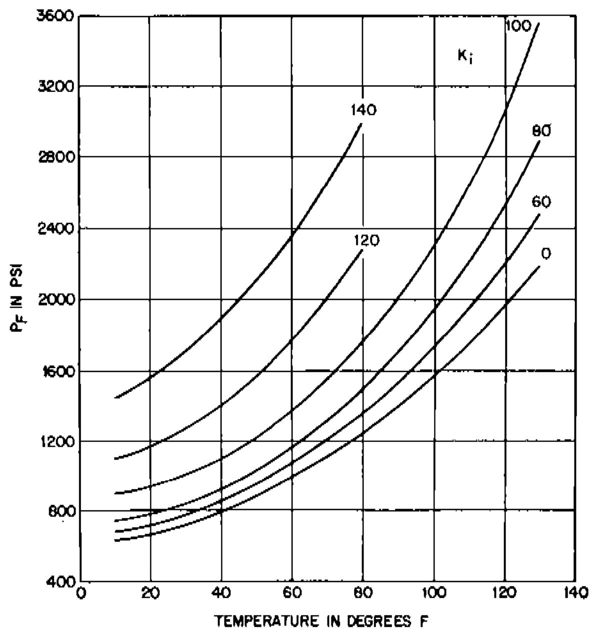

FIGURE 7. Graph of front pressure vs temperature for $K_N = 200$.

80; at 70 and 10 F, they are approximately 1,350 and 760, respectively. The former value is close to that assumed in calculating the burning time at 70 F, so that calculation was correct.

22.6 TUBULAR GRAINS OF FAST-BURNING PROPELLANT

Despite the theoretical advantage of the cylindrical shape, it is preferable in practice to depart slightly from it in the interest of loading convenience. The first rockets developed by CIT had cylindrical grains which were held in the center of the motor tube by plastic tabs cemented to the grain. The extra operation involved in attaching these tabs can be eliminated by extruding the powder with three ridges extending beyond the cylindrical surface and fitting closely the inside diameter of the motor tube. Most CIT rockets use grains of this shape.

If a tubular grain of ballistite is fired statically so that a record of its pressure-time history can be obtained, the curve typically looks like the solid line in Figure 8, with a marked pressure peak near the middle of burning. Numerous experiments[1g,8] have demonstrated that the irregularity of the curves is caused by unstable burning in the central perforation which causes very large stresses on the grain, and that it can be eliminated by at least three devices:

1. Drilling a sufficient number of radial holes through the grain joining the central perforation with the outside (see Figure 9);
2. Inserting in the central perforation a loose-fitting rod of some material which will remain in place throughout the burning;
3. Making the central perforation irregular instead of circular in cross section.[9]

Of these alternatives, only radial holes have been used in service rockets. Rod stabilization has been avoided because it introduces more complexities into the design and the loading than do the radial holes. Noncircular perforations upset the balance between the rates at which the internal area increases and the external area decreases, and hence result in less satisfactory pressure-time curves, a disadvantage which is not serious.

Virtually nothing is known of the mechanism by which these three devices stabilize the burning. Consequently, if one is designing a grain with radial holes, he must determine the optimum number, size, and spacing by trial and error. Numerous experiments at CIT have yielded a number of general rules which are summarized as follows:[1g]

1. Two or more holes in a given plane at right angles to the axis of the grain have no more stabilizing effect than a single hole at the same point.
2. The stabilizing effect of a hole is slightly decreased if the diameter of the hole is made very small in comparison with that of the axial perforation. However, increasing the diameter of the radial hole beyond 0.4 times that of the axial perforation does not add to the stabilizing effect.
3. Critical spacing between radial holes appears to be nearly independent of web thickness and diameter of axial perforation, but is a function of powder composition and increases as heat of explosion and burning rate decrease. Thus for ballistite, the "hottest" powder in general use, the maximum

[redacted]

permissible spacing is about 1 in., while for the slow-burning German propellant it is at least 10 in. so that many grains require no radial holes at all.

4. The exact arrangement and position of the holes is of no importance, except that the stabilizing effect extends only over the region of the radial holes, which must therefore be distributed along the whole length of the grain.

If the web thickness is greater than the radius of the hole, as is always the case, the effect of a radial hole is to increase the surface area at the beginning of burning, since the area of the sides of the little cylinder of powder removed is greater than that of the ends. At the end of burning, the situation is reversed, and the addition of radial holes decreases the surface area. Thus radial holes introduce a regression into an otherwise neutral-burning tubular grain. With the web thicknesses which have usually been used, this effect is not objectionable because it is partially compensated by the tendency of the burning rate to increase as burning proceeds,[1a] so that the resultant regression is slight. The effect becomes large with thick-webbed grains, as is easily seen by consideration of an example, the 2.5-in. by 0.4-in. grain once proposed for the 3.25-in. AR motor.

Outside diameter	2.5 in.
Inside diameter	0.4 in.
Web thickness	1.05 in.
Radial holes	19 holes $\frac{3}{16}$-in. diameter
Length	20 in.
Area without radial holes	182 sq in.
Initial area with radial holes	298 sq in.
Final area with radial holes	137 sq in.
(Final area)/(initial area)	46 per cent

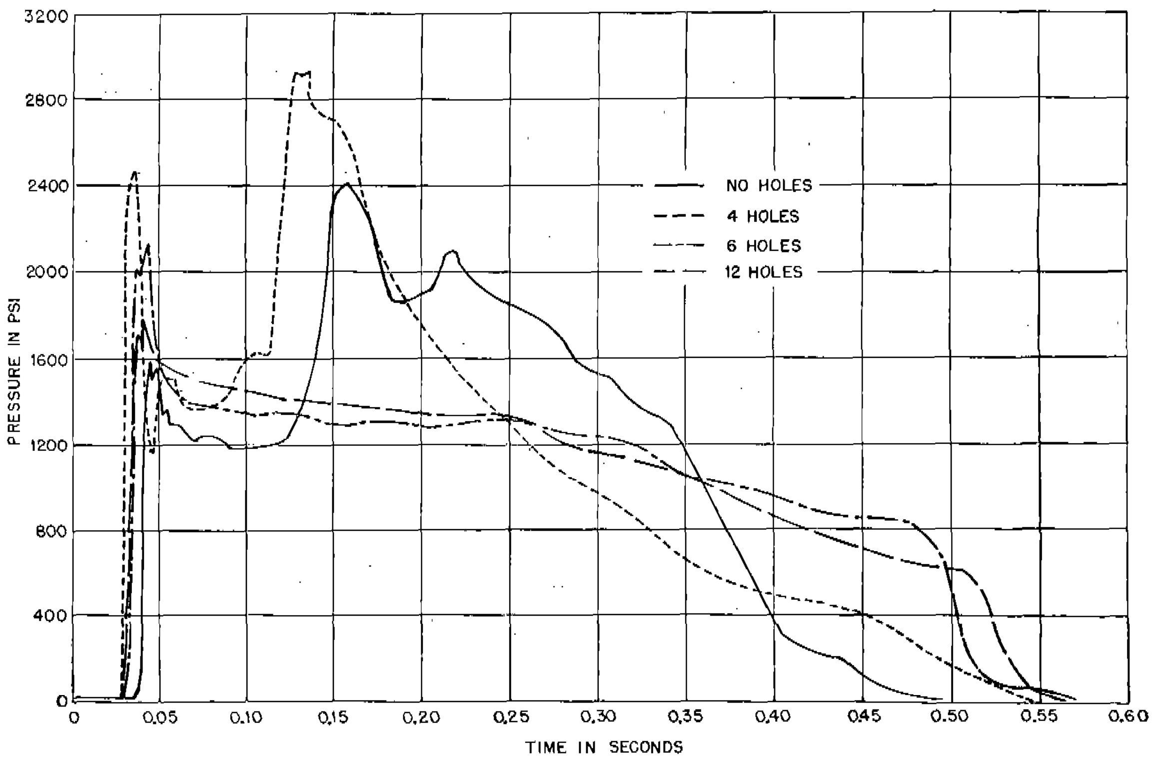

FIGURE 8. Effect of number of radial holes upon performance of tubular 3-ridge grain (1.7 x 0.25-in., 11.4-in. long).

At the end of burning, the $\frac{3}{16}$-in. holes have increased to 1.2 in. in diameter, and nineteen such holes in a 20-in. grain seriously reduce its ability to withstand the acceleration forces.

Tubular grains with radial holes have not, in fact, been used where thick web has been required. A typical tubular grain is the Mk 1, which has been used in the 4.5-in. BR. It is a three-ridge tube 1.7 in. by 0.6 in. in diameter and 11.5 in. long, having twelve $\frac{1}{4}$-in. radial holes and neither end inhibited. Because of the ridges, radial holes, and uninhibited ends, the burning area decreases from an initial value of 98.9 sq in. to a final value of 66.4 sq in., or

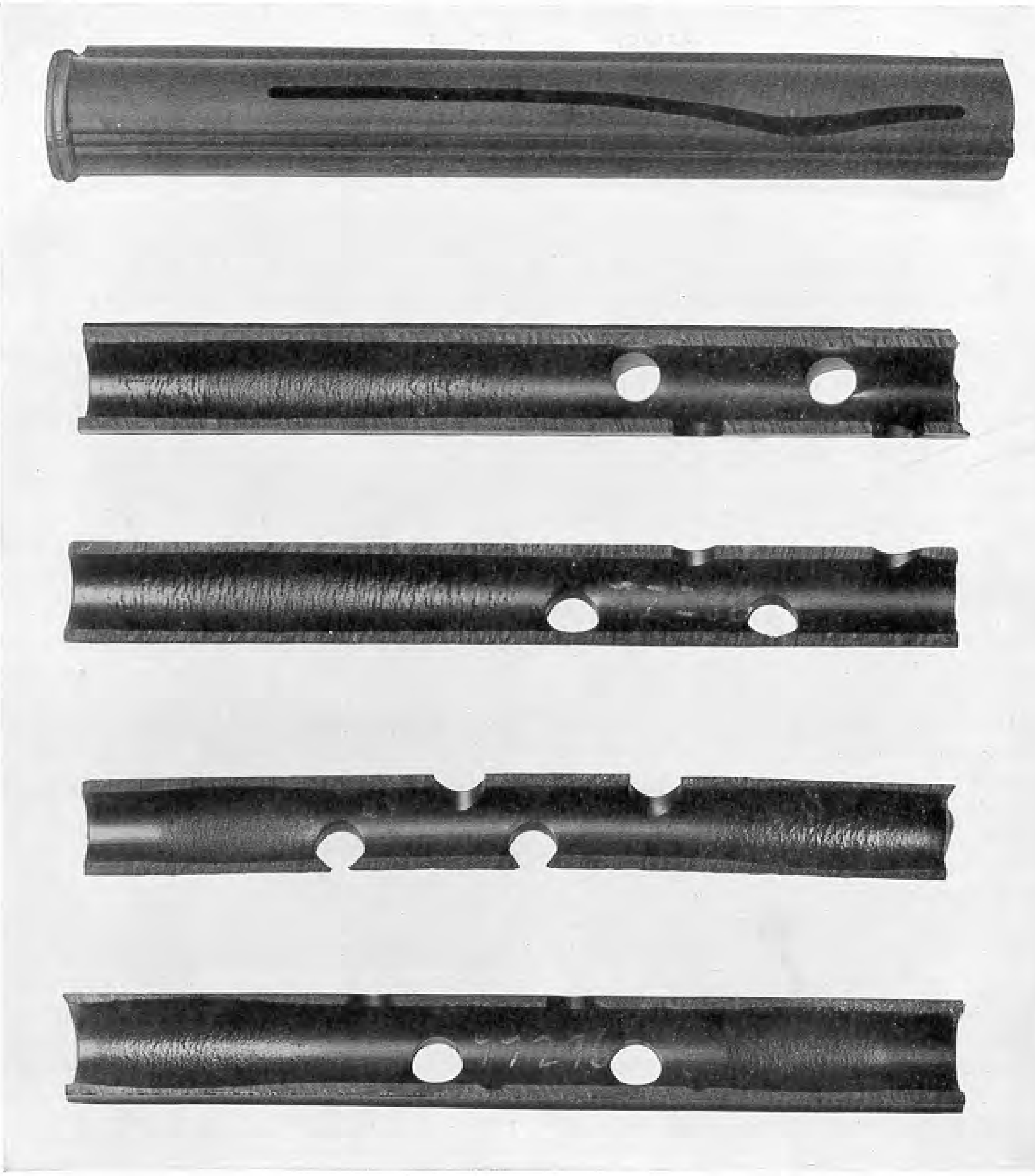

FIGURE 9. Partially burned tubular grains showing stabilizing effect of radial holes. At the top is a 3-ridge undrilled grain in which a large fissure has opened up, causing a sudden increase in burning area and a pressure peak such as is shown in Figure 8. Below are sections of two drilled grains showing that the stabilizing effect of the holes extends only to their immediate vicinity.

33 per cent. Of this, 7 per cent is due to the effect of the ridges, 17 per cent to the radial holes, and 9 per cent to the ends. The change of burning rate as the reaction proceeds introduces a compensation of about 26 per cent, however, so that the pressure-time curves are only very slightly regressive.

[redacted]

22.7 OTHER GRAIN SHAPES

Various shapes other than tubular have been used for propellant grains.[g] These fall generally into four categories: internal-burning, end-burning, external-burning, and multiweb grains. Among the external-burning grains (which, to date, have been the most important) are included slab-shaped, cruciform, and three-, six-, and eight-armed grains. The cruciform is the only one of these now in general use, and, since it illustrates the problems encountered with any exterior-burning grain, we shall confine our discussion to it. CIT experience with the other shapes[h] is summarized in *The Interior Ballistics of Rockets*.[1f]

22.7.1 Cruciform Grains

In any external-burning grain, the area decreases steadily throughout burning, and it is necessary to inhibit the burning of certain portions of the grain if an even approximately neutral burning is to be achieved. The inhibiting process, although costly, time-consuming, and generally a nuisance, has the advantage of providing a large measure of control over the shape of the burning curve. Thus on the cruciform grain, naturally regressive when uninhibited, a very progressive burning curve can be achieved (see Figure 3) by inhibiting the outer cylindrical surface at the ends of the arms along their full length. A neutral burning curve is obtained when approximately 45 per cent of the curved surface of the arms is inhibited in the proper way.[i]

Cruciform grains have been used in preference to tubular (1) when large powder weights have been required as in the aircraft rockets, (2) in spinners where the fact that the inhibited portion remains in contact with the motor wall throughout burning makes them better able to withstand the centrifugal force, and (3) in cases where longer burning times were desired than available cylindrical shapes would provide.

[g] Dimensions of most of the shapes extruded by CIT may be found in reference 10. Complete information on all grains recommended for service use is given in reference 11.

[h] See also the following reports: on 2.74-in. cruciform: references 12 and 13; on 4.2-in. cruciform: reference 14; on hexaform: reference 15; on triform: reference 16.

[i] For reasons which are not understood, the pattern of the inhibiting strips is critical. The effect is probably akin to that which causes unstable burning in tubular grains without radial holes. Experiments with various inhibiting patterns are summarized in *The Interior Ballistics of Rockets*.[1f]

22.7.2 Internal-Burning Grains

A tubular grain which burns only in the central perforation is normally very progressive but can be rendered neutral by putting longitudinal grooves in the central perforation so that it is roughly gear-shaped in cross section. Research on internal-burning grains has been very limited until the last few months, but they hold considerable promise because of the high loading density which they provide and because of the elimination of problems associated with heating of the motor tubes. The inhibiting problem is considerably more severe here than with external-burning grains because of the large surface area which must be inhibited. If the propellant can be molded instead of extruded, a good way to make a charge is to mold it in the motor tube in direct contact with the tube walls.

22.7.3 End-Burning Grains

End-burning grains are similar to internal-burning grains in their design problems. They can be used when extremely long burning times and relatively small thrusts are required. CIT experience with them is summarized in *The Interior Ballistics of Rockets*.[1h]

22.7.4 Multiweb

Two types of multiweb charges were investigated in the early days: two concentric tubular grains and the "4-spoke" or "okra" grains. They were designed primarily to get shorter burning times without sacrificing loading density. No service requirement materialized for such charges and no summary of CIT experience with them exists. They are discussed in a number of reports, however.[17,18]

22.8 LOW-TEMPERATURE PERFORMANCE

As the temperature is decreased, the effective gas velocity of a rocket motor decreases, the reaction pressure decreases, and the burning time becomes correspondingly longer. In some applications these factors may so decrease the accuracy or range that the rocket ceases to be tactically useful, but ordinarily the specified lower temperature limit is determined by the temperature at which the motor ceases to burn continuously.

In the open air, a stick of ballistite will, of course, burn at atmospheric pressure. Inside a motor, however, where no oxygen is present after the propellant gas has swept the air out of the chamber, the chemical reaction involved in combustion is somewhat different and will not proceed if the pressure drops below a minimum which depends on the propellant composition but is usually several hundred pounds per square inch. At lower pressures, the rate of the reaction is so low and the transfer of heat from the gas to the solid grain is so poor that the grain surface is not kept warm enough and burning may stop. When this happens, the partially burned grain can sometimes be recovered, but more often the motor walls, grid, and other metal parts in contact with the grain have been heated enough that they reignite the grain, which burns more or less normally for another period and perhaps again goes out. This process is usually called "chuffing" and motors have been known to "chuff" as many as fifteen times. The time between successive chuffs may vary between half a second and several seconds, and in rare instances periods of more than a minute have elapsed between the first burning period and the first chuff. Chuffing is a serious matter not only because it will cause the rocket to miss its target completely, but especially because the first period of thrust may be just sufficient to free the rocket from the launcher and subsequent chuffs may send it in an unpredictable direction with nearly its normal velocity so that the fuzes may be armed.

Intermittent burning of this type is caused primarily by too low a pressure. Chuffing is associated with low-temperature firing only because sufficiently low pressures are not otherwise obtained normally. Hence the lower temperature limit can be made as low as desired by operating the motor at a high nozzle K so that the pressure is kept up. This can usually be done only at the expense of high-temperature performance, so that the choice of nozzle K depends partly on the relative importance of the two ends of the temperature scale. By the use of a blowout disk (see Chapter 23), it is possible to operate at high K for low temperatures and low K for high temperatures and thus extend the working-temperature range at both ends.

22.9 MOTOR FAILURES AT HIGH TEMPERATURE

22.9.1 Types of Failures

Failures of nonrotating motors at high temperatures are of three principal types.

1. On low-performance motors where the grain is subjected only to small forces, failure may occur because the normal operating pressure of the motor at that temperature is too high for the strength of the metal parts. Since motors are usually designed with a safety factor of 1.5 or 2 at 120 F to 130 F, the temperature required for this type of failure is high—perhaps 160 F to 170 F for ballistite. Failure occurs either at the weakest part of the motor (for example, the nozzle is frequently ejected) or at the front end of the tube where the pressure is highest. It takes place very early in burning before the rocket leaves the launcher.

2. On long-burning-time high-performance rockets, the motor tube may be so weakened by heat that it will burst at relatively low pressure, opening up just ahead of the nozzle, where the heating effect is greatest. Such bursts are not very violent and can only occur at almost the end of burning.

3. Most frequently, motor failures result from collapse of the grain, the sudden increase in burning area sending the pressure skyrocketing. Such bursts are usually extremely violent, at times amounting almost to a detonation, and occur sometimes immediately upon ignition, imparting almost no velocity to the head, sometimes a few feet off the launcher, and sometimes near the end of burning when the web has become thin. Whether the front end or the nozzle end of the tube fails seems to depend on the motor. In many cases the tube opens up along its whole length. Grain failure is the cause of almost all bursts of high-performance motors and a considerable proportion of those of low-performance motors. With some powders which are exceptionally brittle when cold, grain failures may occur at low temperatures. In the category of grain failures are included also the bursts of spinners near the end of burning because of fracture of the propellant by the centrifugal force.

In addition, motor bursts may result from faulty design, such as insufficient radial holes, incorrect inhibiting pattern, or insufficient space at the front end of the motor so that the igniter fractures the grain. We shall consider only the failures of reasonably well-designed motors.

[redacted]

22.9.2 Stresses on Grains

During burning, the propellant grain is subjected to longitudinal compressive stresses from the following sources: (1) difference in pressure between front and nozzle ends of the grain, (2) skin friction between the flowing gas and the grain, (3) impact of the flowing gas on projecting portions of the grain, and (4) acceleration of the rocket. All these types of stresses increase with increasing temperature, types (1) and (4) very markedly. In motors of orthodox design, having the grain supported at the rear end and all the gas flowing toward the rear, all the forces act in the same direction, and the maximum stress in the grain occurs at the nozzle end. The forces are discussed in greater detail in *The Interior Ballistics of Rockets*,[1] from which Table 2 and Figure 10 are taken, and in reference 19. These figures represent conditions for the Mk 13 grain used in the 3.25-in. AR motor. The discontinuities in the curve of Figure 10 are due to the localized impact forces on projecting portions of the grain at the front end of the inhibitor strips (see Figure 11). It is seen that the most important forces are the pressure differential and the acceleration. To reduce the former, it is necessary to reduce K_I, but, for a given propellant shape, K_I is proportional to the grain weight and in high-performance motors is necessarily large. Unfortunately, K_I increases slightly with increasing temperature because the powder has a larger coefficient of expansion than the motor. High acceleration also is usually associated with high-performance motors, so that obtaining good performance at high temperatures is the most difficult problem in designing such a motor.

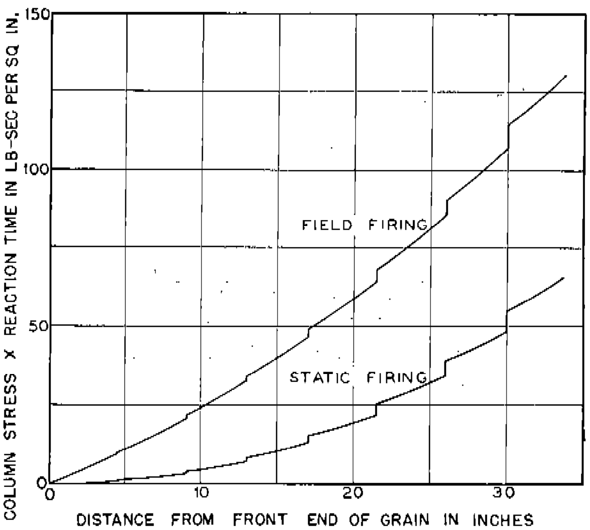

FIGURE 10. Compressive stress in Mk 13 cruciform grain, 50 per cent burnt.

TABLE 2. Total compressive stress acting on Mk 13 propellant grain in firing at 140 F.

Per cent of grain burned	Total stress (psi)	
	Static	Flight
0	323	441
10	246	364
29	164	282
48	119	237
67	96	214
86	90	208
91	101	219
95	127	245

22.9.3 Mechanism of Failure

The mechanism by which grain failures occur is discussed in *The Interior Ballistics of Rockets*.[11] When burning starts, the compressive stresses on the grain cause it to become shorter and fatter, bulging particularly at the nozzle end where the stresses are greatest. The bulging decreases the port area around the grain and hence causes the pressure drop to become still larger. The amount of bulging is determined by the elastic modulus and Poisson's ratio for the propellant. A "strong" grain will bulge only slightly and equilibrium will be reached at a higher pressure than normal because of the higher K_I;[j] but a "weak" propellant will bulge so much that an unstable condition results, higher front end pressure causing more bulging which in turn causes still higher pressure. Thus the effect is virtually as if the nozzle were closed, and the pressure quickly builds up to a value which will fracture the grain and burst the motor. It can be seen from this analysis that the ultimate compressive strength of the propellant is of secondary importance, but the elastic modulus and Poisson's ratio primarily determine the minimum value of the pressure drop at which instability begins.[k]

At the end of burning, a grain fails because it becomes too slender relative to its length to with-

[j] There is some evidence that oscillations of the grain about its new equilibrium shape can occur.[20]

[k] Tests of the strength of various propellants are summarized in references 21–24.

[redacted]

stand the forces—principally the acceleration force —and buckling occurs. Usually the collapse does not burst the motor but merely puts a sharp peak at the end of the pressure-time curve. Increasing the acceleration moves the peak closer to the beginning of burning so that more powder is ejected unburnt and the effective gas velocity decreases. One important reason for the better performance of cruciform than tubular grains in high-performance motors is the fact that the inhibited portions of the former continue to be supported by the motor walls throughout burning.

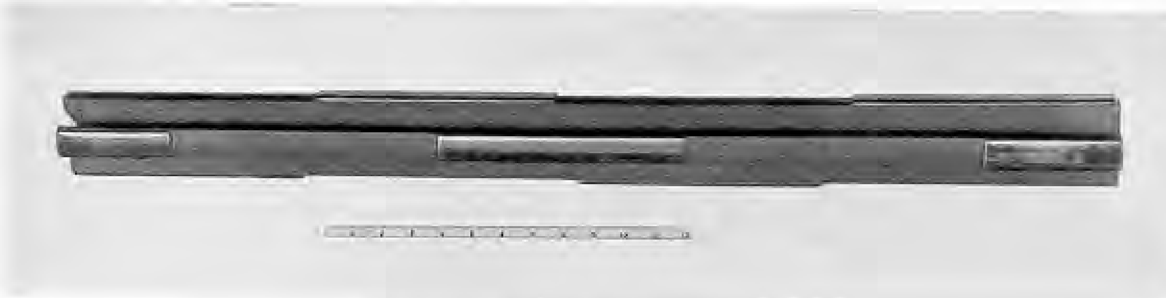

FIGURE 11. Mk 13 cruciform grain, inhibited.

In spinner motors, the centrifugal forces can also induce grain fracture and burst the tube, and it is frequently this consideration which limits the spin velocity. Two types of failures have been observed. At high temperatures the spin is increased because of the increase in effective gas velocity, and bursts occur near the end of burning at a temperature which depends principally on the tensile strength of the propellant. Bursts may also occur at very low temperatures, and here they seem to be associated not with low tensile strength but with brittleness of the propellant.

22.10 DESIGN OF MAXIMUM WEIGHT GRAINS

It is frequently required to design a motor which contains the maximum possible amount of propellant for the given caliber either with a specified web thickness or regardless of web thickness. The most important factor limiting the powder weight is the internal K. We have already seen that large K_I means large forces on the grain, but another effect is involved which is perhaps equally important. Whenever the gas flows at high velocity over part of the burning surface, the burning rate in these regions is considerably accelerated, probably because such conditions are conducive to more rapid transfer of heat from the gas to the grain. This effect is termed "erosive burning." Since the gas velocity is always higher at the nozzle end, the burning tends to be faster at this end, but the effect is compensated by the pressure dependence of the burning rate which tends to make burning faster at the front. If the ratio K_I/K_N is too high, however, the erosion becomes very severe so that the decreased web thickness at the nozzle end contributes to the grain collapse. In practice it has been found that values of K_I/K_N greater than about 0.75 cannot be used, and for service rockets it has not been felt desirable to exceed 0.60. For CIT rockets, which have K_N usually in the neighborhood of 200, this sets the upper limit of K_I at 120. Only on the 3.25-in. Motor Mk 6 was this value exceeded, and this was one of the reasons for abandoning it in favor of the Mk 7.

With the stipulation that the internal K cannot exceed a certain value, it is simple to plot curves of the relations between external diameter, web thickness, length, and K_I from which the maximum possible tubular grain for a given motor caliber can be obtained. The radial holes and supporting ridges complicate the calculations considerably and hence are usually omitted from the calculations, so that the curves can be used to show qualitative relations only. A series of such curves showing the effect on the internal K of a constant weight grain caused by varying the inside diameter, outside diameter, average diameter, or web thickness is given in *The Interior Ballistics of Rockets*.[11]

If only the restriction on internal K is considered, one finds that the maximum weight tubular grain for a given motor diameter is obtained by making the axial perforation as small as possible, the outside diameter less than 0.6 times the internal diameter of the tube, and the length of the grain approximately 60 times its diameter. Such grains are not practicable, however, for a number of reasons. Grains with extremely small axial perforations cannot be made because the slender "stake" required would not withstand the forces encountered in extrusion without wandering. The lower limit is in the neighborhood of 0.1 times the motor internal diameter. Very long grains are not used because of the excessive weight of motor tube required to encase them and because they are obviously ill adapted to stand up without buckling under the high longitudinal acceleration forces encountered during firing.

Empirically, it has been found that the heaviest tubular three-ridge grain of JPN ballistite which will perform satisfactorily at 140 F statically and 130 F in the field has the following characteristics:

Outside diameter	0.83 calibers
Inside diameter	0.13 calibers
Web thickness	0.30 calibers
Length	6.64 calibers

Here the "caliber" is used as a unit of length equal to the internal diameter of the motor tubing. This unit is used in nearly all discussions of grain size because it enables the results to be expressed in a form independent of the actual size of the motor. The maximum grain dimensions tabulated above have been found to be correct for 2-in. and 3-in. calibers,[l] and it appears likely that they would be approximately correct for any caliber. Thus for a 4.625-in. motor like the HVAR, the maximum tubular grain would weigh slightly less than 20 lb.

A very useful method of representing the relation between grain shapes and weights is that adopted in reference 25 [m] and in Figure 12. Except for the single curve marked "cruciform," all the data in the table are for a single tubular grain inhibited on both ends and having no radial holes. If one writes down expressions for the volume, burning area, and port area of a grain and calculates the possible powder volume corresponding to a particular value of K_I, it is immediately apparent that all linear dimensions are proportional to the internal diameter of the motor tube and the volume is thus proportional to its cube. Consequently, one can draw a set of curves (dashed lines in Figure 12) giving the relations between length, volume, and web thickness (expressed in terms of dimensionless parameters) of grains having a particular K_I, and the curves will apply to all motors and every charge whose web thickness is everywhere the same and whose burning surface remains constant during combustion. The restriction that these grains have dimensions which allow them to fit into the motor tube limits us to particular portions of the curves showing volume (or weight) as a function of web thickness, the allowable region depending on the type of charge. Thus a tubular grain cannot have a diameter larger than that of the motor nor an axial perforation smaller than zero, so that, unless we are willing to work at a different value of internal K, we cannot use a grain of dimensions corresponding to a point in Figure 12 outside the area bounded by the curves "MAX OD" and "ID = 0." In practice, of course, one must remain within a somewhat more confined region, and it has been CIT's practice to use outside diameters only between 0.8 and 0.9 times the inside diameter of the tube and thus keep the grains short. It is shown in reference 25 that all charges consisting of combinations of more than one tubular grain have maximum volumes less than that obtainable with a single tubular grain. In fact, it is easy to see that no other grain shape can approach the single tubular grain in possible loading density if only geometrical factors are considered.

We have seen that, in practice, the cruciform shape gives the highest loading density, and it is of interest to show its characteristics on the same graph with the tubular grains. One less variable parameter is available with cruciform grains than with tubular, so that a single curve is obtained instead of a permissible region of the graph. Plotted in Figure 12 is such a curve which assumes (in accordance with CIT practice) that the outside diameter of the powder is 0.91 times the tubing inside diameter (to allow for the inhibitor) and that 45 per cent of the cylindrical surface and both the ends are inhibited. The curve shows that the use of cruciform does not allow us to get more powder

[l] The heaviest 2-in. grain which has been used is the Mk 16, which has dimensions 1.7 x 0.28 x 12.5 in., weighs 1.75 lb, and is used in the 2.25-in. subcaliber aircraft rocket. Its weight and web thickness are plotted in Figure 12. If this grain is scaled up by the factor 1.5 appropriate to a 3.0-in. ID motor, it becomes 2.5 x 0.4 x 18.8 in. and weighs 5.9 lb. This is only slightly shorter than the longest tubular grain which would function in the 3.25-in. AR motor having an internal diameter of 3.01 in.

[m] Figure 12 is copied from this report except for the curve on cruciform grains, which had not previously been published.

[redacted]

in a given length.[n] Its sole advantage is that longer grains of this shape can be made to perform satisfactorily because (1) the inhibited outer surfaces are supported by the motor tube throughout burning, (2) grains do not need to be weakened by radial holes as do tubular ballistite grains, and (3) the inhibitors reduce the surface to volume ratio, decreasing the K_I per pound of propellant.

Also plotted on the graph are points corresponding to the heaviest tubular and cruciform grains which CIT has found practicable.[o] It is possible to conclude from the data that the maximum ballistite grain which can be put into a motor tube having an internal diameter of D in. without exceeding $K_I = 120$ is approximately $0.22\ D^3$ for tubular grains and $0.33\ D^3$ for cruciform grains. These values may be useful as a rough empirical rule. If lower K_I is desired, the maximum weight is reduced proportionately.

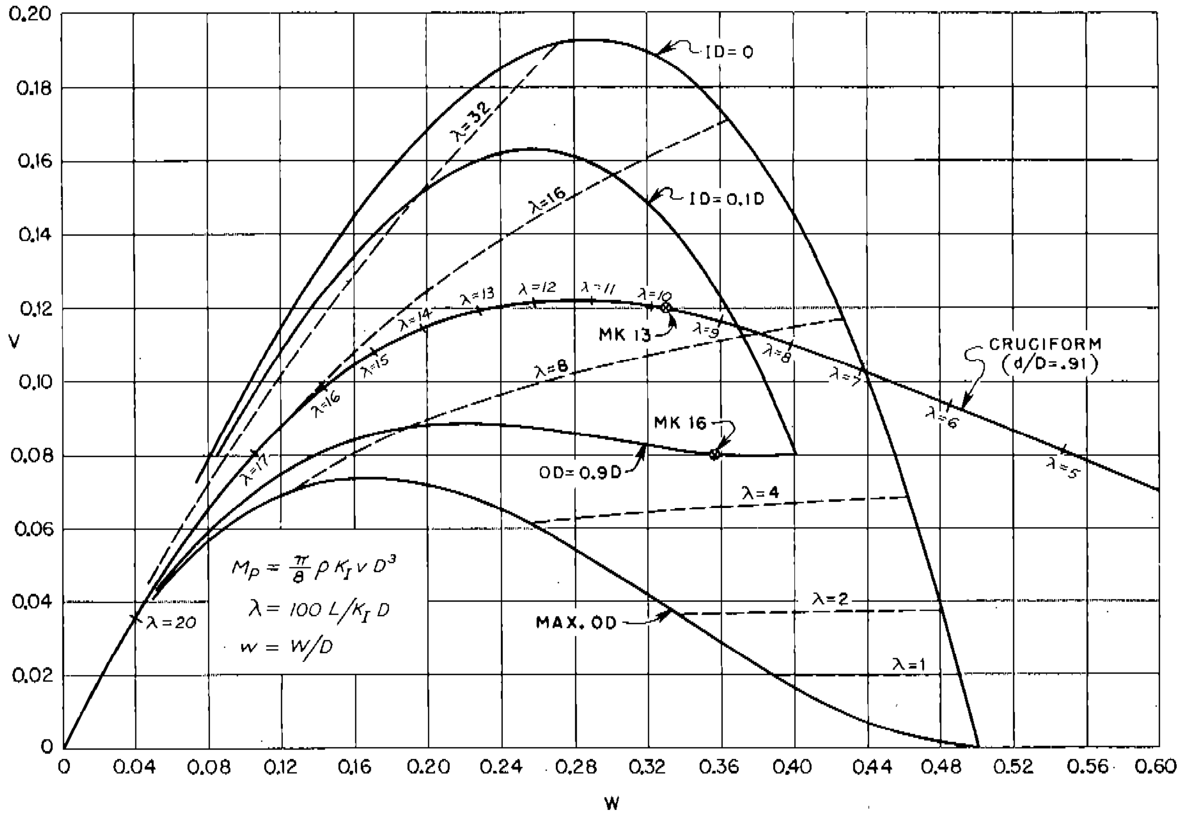

FIGURE 12. Theoretical maximum volume of cruciform and tubular grains as a function of web thickness (see Table 1 for definition of symbols).

22.11 IGNITERS

The function of an igniter is twofold: to heat the propellant grain to ignition temperature and to bring the pressure in the motor up to a point where grain will continue to burn satisfactorily. It must accomplish these purposes with a short and reproducible delay[p] at all temperatures at which the rocket is to be used and must not subject the grain to excessive forces when it ignites. For efficient heat transfer to the grain, it is desirable that the products of combustion of the igniter include an appreciable amount of solids, since the radiation from gases is relatively low. At the same time, however, some

[n] That the values of λ for cruciform and tubular grains are not identical results from the fact that the cruciform web thickness is not strictly uniform, and a sliver is left after burning.

[o] The points were plotted in their proper place with respect to web thickness and weight (w and v). That the values of OD and λ read from the graph are not quite correct results from the simplifying assumptions made in plotting the curves.

[p] Short ignition delays are obviously of special importance in aircraft rockets.

[redacted]

gaseous products are necessary to increase the motor pressure rapidly to the desired value.

All CIT rockets are ignited electrically. In a typical igniter, a squib, approximately ¼ in. in diameter and ½ in. long, which consists of a clay body with a small depression at one end containing an electric bridge wire and a heat-sensitive explosive material, is placed in contact with the main ignition charge. Originally a booster charge of finely divided black powder or flash powder was placed between the squib and the main charge, but it was found to be unnecessary in properly made igniters.

Black powder, usually in the FFFG granulation, has been used almost exclusively for the main ignition charge. Ballistite turnings have some advantages over black powder for static-firing tests, but give long ignition delays.[26] Tests of igniters containing mixtures of magnesium and potassium perchlorate [27] showed them to be distinctly superior to black powder igniters giving shorter delays at low temperatures and lower pressures at high temperatures. They have not been used in service rockets, however, because they are thought to be more hazardous than black powder igniters and because the magnesium is very subject to surface oxidation during storage, with resultant deterioration in performance. Tests of magnesium-potassium perchlorate mixtures as a squib booster for black powder igniters showed a negligible improvement in performance.[28]

FIGURE 13. Igniter types. (A) 2-in. brass case with bakelite closure, (B) 3-in. plastic case, (C) metal case for Tiny Tim (230 g capacity), (D) HVAR metal case showing inside with clips for holding squibs and bottom and top views of assembled igniter. (Note that relative sizes are not accurately shown.)

The factors involved in making good igniters are discussed in more detail in *The Interior Ballistics of Rockets*.[1j] For short ignition delays, it is required that (1) the firing current be above a certain minimum (about 2 amperes in CIT igniters) so that the bridge wire is heated quickly; (2) the ignition charge be tightly compacted; and (3) the igniter case be strong enough to remain intact until all parts of the

[redacted]

charge have ignited. In the latter regard, considerable care is necessary, since a strong igniter case will give a high pressure peak at ignition, thus reducing the safety factor of the motor and contributing to high-temperature failures, and it may also burst with such violence as to fracture the grain. These problems are not serious in fin-stabilized rockets where there is usually adequate space at the front end of the motor to cushion the shock of the igniter's burst. In spinners, however, where length is at a premium, the strength of the igniter case is very critical, and one may be forced to accept a slight increase in ignition delay in order to prevent grain fracture.

Other desirable igniter characteristics include ease of fabrication and loading, ruggedness and resistance to vibration, watertightness, and the property of fragmenting in such a way as to leave no pieces large enough to obstruct the nozzles. The types of igniters which have been used in service rockets are shown in Figure 13 and discussed briefly below.

The earliest CIT service rockets contained brass can igniters with bakelite closures.[29] A drawn brass can containing the powder and the squib was crimped over a close-fitting bakelite disk which was perforated for the squib leads. The crimping operation compacted the powder to the desired degree, and the igniter was reasonably sturdy. Its disadvantages were frequent squib breakage and poor resistance to moisture, and it is now considered obsolete except for experimental work.

Igniter cases of molded plastic have been used extensively.[1j,30–34] They provide good support for the fragile squib by enclosing it in a special compartment and, having threaded closures, allow the charge to be very firmly compacted so that their resistance to impact and vibration is very good. They can withstand complete immersion in water for several days. For single-nozzle ground-fired motors of 2-in. and 3-in. caliber, they are completely satisfactory. For smaller motors and spinners, however, they cannot be used because the cases, in order to be sufficiently strong, must have walls approximately 0.1 in. thick with numerous reinforcing ribs of greater thickness, so that relatively large fragments are produced when the case breaks up, and these may plug the nozzles. The squib compartment is especially bad in this regard since it is thick and usually remains intact. In a single-nozzle motor, a plugged nozzle means a motor burst. In spinners, the primary effect is a decrease in accuracy, although bursts may result in extreme cases. Finners larger than 1.25 in. in diameter have not been made with nozzles small enough to be plugged by igniter fragments, but even here the fragments may be a disadvantage since they are ejected through the nozzles at high velocity and may damage the tail surfaces, radiators, etc., of aircraft. In order that the case may open up at pressures small enough to do no damage to the grain, the closure must be made with internal threads on the case.

Igniters with metal cases can be made fully as waterproof as those of molded plastic, are even more resistant to mechanical stresses, are not affected by nitroglycerin (as are some plastics), do not break into large fragments when fired, and are especially cheap and simple to make. The cases have been made of 0.010-in. tin-plated steel, which is the same weight as the material for ordinary tin cans; in fact, standard sizes of commercial cans can sometimes be used. In igniters for large rocket motors, it has been the practice to include two squibs wired in parallel, thus considerably reducing the number of misfires since squib failure is their most important cause.

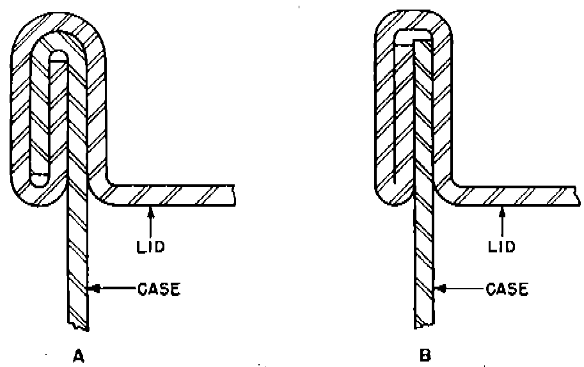

FIGURE 14. Crimps for metal case igniters: (A) standard double crimp, (B) false crimp.

For finned motors, tin plate igniter cases have been made with the standard double crimp (A of Figure 14) which is used for commercial cans. This crimp is strong and requires considerably pressure inside the case before it fails, thus giving the short and reproducible ignition delays which are essential for aircraft rockets. Although they burst with some violence, such igniters do not injure the grain because of the cushioning effect of the free volume at the front end of the motor. Spinners, having less

[redacted]

than 1 in. between the front end of the grain and the base of the head, require an igniter case which opens up at much lower pressures, and for these the so-called false crimp (B of Figure 14) has been used.

Usually it is desirable to place the igniter at the front end of the motor so that the products of its combustion come in contact with the full length of the grain. For spinners, which are necessarily short and fat, rear end initiation is possible and desirable because the igniter can be put into a space which would not otherwise be occupied and need not subtract from the grain length. Design and tests of a toroidal igniter to fit around the grid stool in the 3.5-in. spinner are discussed in reference 35. It was made from plastic, but not being in contact with the grain or subject to any compression, its walls could be made very thin so that no nozzle-plugging fragments were produced. The igniter worked successfully, but it was developed too late for service use.

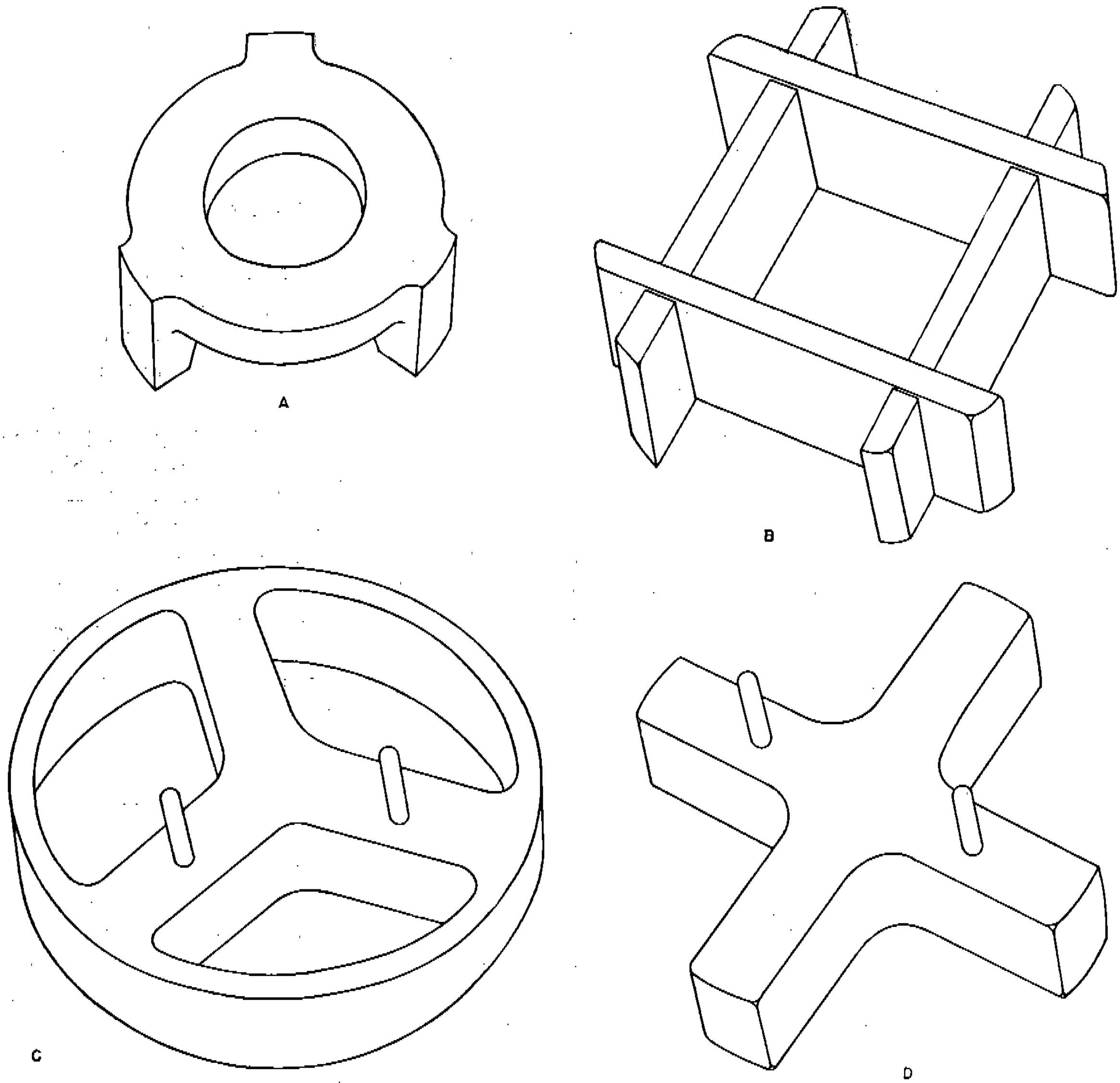

FIGURE 15. Grid types for single-nozzle motors: (A) stool, (B) box, (C) triform, (D) cruciform.

The size of an igniter charge must be determined empirically. Too small an igniter will not give reliable ignition at low temperature, and too large a one will raise the pressure considerably above the equilibrium pressure at high temperatures and thus contribute to motor bursts. With the 11.75-in. motor, a special igniter problem arose in connection

[redacted]

with the shock wave effect on the aircraft structures and caused a drastic reduction in the size of the igniter charge. (See Section 19.5.2.)

22.12 DESICCANT BAGS

Because ballistite is hygroscopic and its burning characteristics are dependent on its moisture content, the practice of inserting a small bag of silica gel in the nozzle exit cone was adopted for the first service motor, the ASR, and became standard procedure. For small motors the efficacy of such a desiccant bag is doubtful at best, and for motors where the propellant weight is several pounds the bag certainly does no good. In the latter case, the moisture capacity of the propellant exceeds that of any desiccant bag of practicable size, so that the only way to be sure that the propellant has the proper moisture content is to load it into the motor when its moisture content is correct and then seal the motor securely so that it cannot change. Thus, if the seals hold, the desiccant is unnecessary, and, if the seals spring a leak, the desiccant is not likely to be equal to the task of keeping the powder dry.

With the advent of multiple-nozzle motors, in which there was no convenient place to put it, the desiccant bag was abandoned. Probably it could also be dispensed with on most of the 3.25-in. motors.

22.13 GRIDS

The purpose of the grid is to support the grain and allow free access of the gas to the nozzle. Its shape is practically dictated by the shapes of the grain itself and of the nozzle, and the principal problem in its design is to make it of the right thickness so that it will not be too heavy or obstruct the gas flow too much and still be strong enough at the end of burning to withstand the forces on it despite the considerable heating and erosion to which it is subject.

Typical grid shapes which have been used are shown in Figure 15. The three-legged (or sometimes four-legged) stool type and the so-called box grid (A and B in Figure 15) have been used for tubular grains. The former can be made of cast iron or cast steel, but must still be machined on the front and rear surfaces and the outer diameter. Although the stool grid offers better support for the grain, the box has usually been preferred because it cannot be put in upside down.[q] Box grids [36] have been made by stamping the pieces from steel strip, assembling them, and machining to diameter. They can also be cast or sintered. The sintering method offers the important advantage that no machining is necessary, but the pieces usually have low density, low strength, and extremely little erosion resistance, so that rigid inspection of them is necessary.

Grids for triform and cruciform grains in 3.25-in. motors are shown in C and D of Figure 15. In the case of external-burning grains of this type, the grid must perform the additional function of keeping the grain from rotating and thus closing up the port area. The most successful method that has been found for assuring this is to have two steel pins projecting above the surface of the grid and indexing into holes in the grain and its plastic end washer (see Figure 16). In cases where it is convenient for loading, the grid has been cemented to the end washer as an added precaution. The most satisfactory production methods for making cruciform and triform grids have been steel casting, torch cutting from plate, and copper-brazing laminations stamped from 11-gauge sheet.

FIGURE 16. End view of Mk 13 grain showing wells in end washer to accommodate grid pins.

More complicated grids are required for multiple-grain motors such as Tiny Tim.

[q] At one time early in World War II, the discovery of one ASR motor with an upside-down grid impelled the X-raying of several thousand motors.

Chapter 23

MOTOR DESIGN

By *C. W. Snyder*

23.1 INTRODUCTION

In this chapter we shall discuss the problems encountered in designing the various components of a rocket motor and the solutions for them which have been used at CIT. Two cautions which apply throughout this book should perhaps be specially emphasized here. The rocket motors in which we have been interested have all been of a special and very similar type, namely, those having pressures seldom out of the range 1,000 to 2,000 psi at ordinary temperatures, burning times in the range 0.2 to 1.5 seconds, and velocities either subsonic or only slightly supersonic. Hence the solutions which we have found must not be thought automatically to apply to rockets differing too much from these specifications. Second, we must ask to be judged, in many cases, by our words and not by our deeds as embodied in service rockets, since far too often important design features were settled by expediency in the war situation rather than by the ideal and, perhaps less frequently, they were settled on the basis of preliminary information which did not prove finally to be correct.

23.2 TUBES

23.2.1 Tubing Dimensions

The size of a motor tube is ordinarily determined by one or both of the following considerations: it must fit a particular propellant grain, thus determining its length and inside diameter; it must fit a particular head, a consideration which, if it exists, usually determines the outside diameter, at least approximately. The accuracy required on any of these three dimensions is never very great. The inside diameter must fit the grain, but the accuracy with which grains can be made in practice is usually less than the commercial tolerance on tubing diameter, especially when the tubing is made to an ID specification, and clearances in the neighborhood of $\frac{1}{64}$ in. on the radius are not objectionable. The tubing diameters become particularly critical only when peripheral variations in wall thickness cannot be tolerated either because of weakening the tube, as in the case of an ultrahigh-performance motor where the absolute minimum wall thickness is being used to save weight, or because of the unbalance introduced thereby, as in a low-dispersion spinner. In either of these cases, the OD, and perhaps also the ID, must be machined since concentricity tolerances, particularly on seamless tubing, are always rather large.

23.2.2 Tubing Material

For tubing material, nothing other than steel has been given serious consideration since alternatives which can begin to compete in price and abundance do not have the requisite strength and high melting point. Seamless tubing is definitely preferred because there appears to be no simple and foolproof method for detecting a defective weld in a motor tube—except possibly by fabricating it into a rocket motor and firing it. This difficulty with welded tubing was most troublesome with the 3.25-in. AR motors, more than half a dozen of which opened up at the seam during high-temperature firing, even though they had all been hydrostatically tested at 4,000 psi and the pressure during firing apparently did not reach half this value. It seems probable that this is to be explained by the more sudden application of the pressure during firing, since the motors burst before having time to get warm.

23.2.3 Wall Thickness

For calculation of the wall thickness and tensile strength required, Barlow's formula is adequate, as the wall thickness is always small relative to the diameter and great accuracy is not required because of the large safety factor which is included. This formula is

$$t = \frac{DP}{2S}, \tag{1}$$

 [redacted]

where t and D are the wall thickness and the outside diameter in inches, P and S are the internal pressure and permissible tensile stress in pounds per square inch. It has been the practice to subject each length of motor tubing or each completed motor to a hydrostatic test at a pressure exceeding the maximum normal pressure at the upper service temperature limit by a factor 1.5 for ground rockets or 2.0 for aircraft rockets. (With improvements in rocket propellants, such large safety factors may no longer be justifiable.) The pressure which must be used in Barlow's formula is thus not the motor pressure but the test pressure. The tensile strength to be used is the yield strength rather than the ultimate strength, since a motor is not considered to have passed the pressure test if it swells by more than a specified amount.

Specification of the test pressure and the motor caliber determine, by Barlow's formula, only the product of wall thickness by yield strength. For a low-performance motor, because the weight is not critical, one usually plans to use ordinary cold-rolled steel tubing, for which 50,000 psi is a reasonable value of tensile strength, and make the wall thick enough to stand the pressure. If high performance is required, one usually prefers to use the highest grade of heat-treated steel available and make the wall as thin as possible to save weight. How far one can go in this direction depends upon the heating effect.

23.2.4 Heating Problems

As the propellant gas flows past any part of the rocket it will transfer heat to the surface mainly by conduction and convection. Heat will also be transferred by radiation, but with ballistite and other smokeless propellants the radiative transfer is so small a portion of the total heat transfer that it can be neglected. The temperature reached by the surface depends on the rapidity with which the heat received from the gas is distributed by conduction throughout the volume of the solid. Ultimately an equilibrium would be established in which the rate of heat transfer to each unit area of the surface would equal the rate of conduction away, but with the short burning times characteristic of CIT rockets, we have to do with transient conditions. The theory of heat transfer and conduction is applied to rocket motors in *The Interior Ballistics of Rockets*,[1a] and in reference 2. We shall consider only the results here.

The time rate of heat transfer from the gas to the rocket's inner surface is proportional to the difference in temperature between the gas and the metal and to the heat transfer coefficient h. The transfer coefficient is very nearly proportional to the "mass velocity" G, defined as the mass of gas

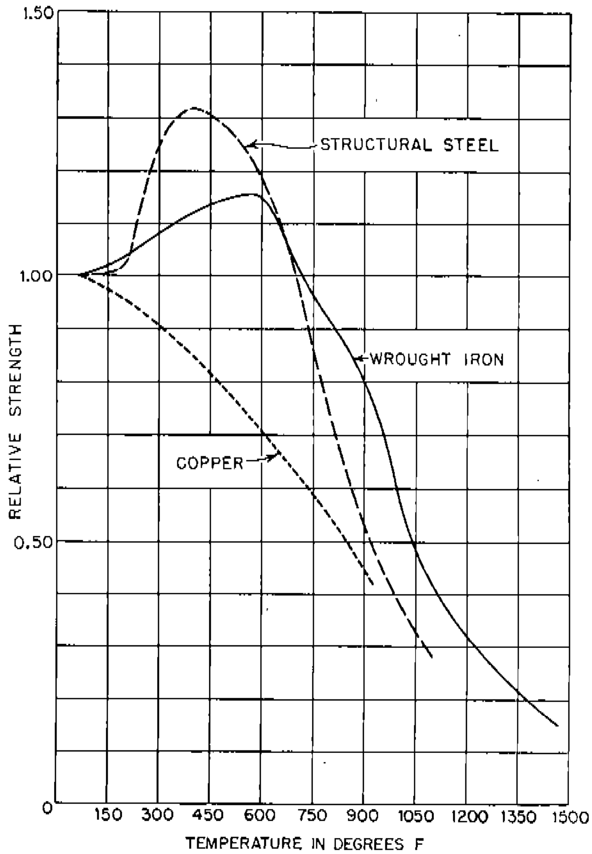

FIGURE 1. Tensile strength of metals at various temperatures.

flowing in 1 second through unit area normal to the direction of flow.[a] Thus the heat transfer during burning to a unit area is least at the front end of the motor, where G is small because gas is practically stagnant, and is greatest at the nozzle throat and at the nozzle end of the grain, where G is greatest because the port area is small and all the gas passes by. During burning, the mass velocity at the nozzle end of the grain decreases rapidly because of the increasing port area, and the rate of heat transfer

[a] Actually it depends on the 0.8 power of the mass velocity through a proportionality factor which is slightly greater for small gas flow channels than for large.

[redacted]

to the surface decreases in proportion. Hence the inside surface of the tube reaches its maximum temperature during the first half of the burning time and remains nearly constant thereafter, but the average temperature of the wall, which is important from the standpoint of strength, increases steadily throughout burning.

The variation of tensile strength with temperature for typical metals is plotted in Figure 1, taken from reference 3. Since the average temperature of a motor tube increases steadily during burning, a curve of burst strength as a function of time during burning would have a very similar shape, with the scale depending upon the type of steel, the wall thickness, and the mass velocity. If the burning time is long enough, the burst strength will eventually fall below the motor pressure, and the motor will fail. It is thus of considerable importance in design to be able to predict the average temperature of the critical point of the motor wall at the end of burning. By using a value of h which has been determined experimentally from a similar rocket, this can be done with considerable accuracy, but the method is involved and will not be given here.[4,5] Typical results of such calculations are given in Tables 1 and 2.

TABLE 1. Effect of firing temperature on heating of 11.75-in. rocket motor wall.

Wall thickness, 0.280 in.		
Firing temperature (°F)	−10	140
Average reaction pressure (psi)	960	1,900
Reaction time (seconds)	1.40	0.70
Calculated metal temperatures at nozzle end of grain at end of reaction (°F)		
Inner wall surface	2000	2400
Outer wall surface	440	300
Average	1040	1000
Average temperature rise (°F)	1050	860
Total heat transferred to wall (Btu per sq ft)	1560	1280

The total amount of heat transferred to the motor wall is slightly greater at low powder temperatures than at high because the decrease in the rate of heat transfer is more than compensated by the greater burning time. At a given powder temperature, the total transfer depends very little on the thickness of the motor wall. Consequently, as illustrated in Table 2, the temperature reached by the motor tube is considerably greater for thin-walled tubes than for thick, because of the smaller

TABLE 2. Calculated temperature distribution in motor wall of 5.0-in. rocket motors at nozzle end of grain.*

Type of motor	Mk 2	Thin-walled	Thin-walled with refractory
Wall thickness (in.)	0.188	0.120	0.120
Refractory thickness (in.)	....	....	0.010
Temperatures at end of reaction (°F)			
Inner refractory surface	....	....	3250
Inner metal surface	2000	2150	900
Outer metal surface	700	1550	500
Average metal	1200	1770	650
Total heat transferred to wall (Btu per sq ft)	1050	1040	330

* Assumed properties:

	k (Btu/ft · hr · °F)	C (Btu/lb · °F)	ρ (lb/ft³)
Steel	25	0.13	490
Refractory	0.6	0.2	160

heat capacity of the thin wall. Whether it is possible to achieve a significant saving in weight by using high-tensile steel and thin-walled tubes depends very markedly on the heating effect. Thus, in the example of Table 2, it would probably not be possible because at 1770 F no steel would have any appreciable strength.

It should be noted that for a given burning time and mass velocity of gas, the absolute thickness of the wall, and not the ratio of the wall thickness to the diameter of the motor, is the determining factor in establishing the temperature. The strength of the motor with respect to internal pressure, on the other hand, depends upon the ratio of wall thickness to diameter. Therefore, in small-diameter motors of fairly long burning times, the minimum wall thickness is generally determined as much by the heating effect as by strength requirements so that material of unusually high tensile strength offers no great advantage. With the larger units such as the 11.75-in. motor, a wall thick enough to have adequate cold strength is of ample thickness to keep the temperature within reasonable limits, and considerable weight reduction can be made by using high-strength steels.

23.2.5 Refractory

The amount of heat which the steel wall must absorb can be much reduced by insulating it with a thin layer of refractory material. A typical refractory may have about one-fortieth the heat conductivity of steel, so that, ideally, the addition of a very thin layer of refractory on the inside would

[redacted]

reduce the wall temperature as much as would a considerable increase in wall thickness, with its attendant weight increase. Practically, however, the low conductivity of the refractory causes its inner surface to approach the temperature of the gas, and its relatively low tensile strength causes it to be eroded away fairly rapidly, so that its effectiveness does not approach the theoretical value. It is possible, nevertheless, to achieve a significant saving in weight by the use of refractory and high-strength thin-walled tubes, as is shown by tests at CIT[2] and by the British experience with the RP–3. It was not felt that, in the tactical situations for which CIT's rockets were developed, the increase in performance attainable by refractory coatings justified the increased complexity of manufacturing. Hence no very extensive investigation of refractories was made. In the future their use may be desirable and will change many of the conclusions in this book.

23.2.6 Internal-Burning Grains

It should be noted that, throughout the discussion of the heating effect, it has been assumed that the propellant burns on the outer surface so that the gas is in contact with the wall. Near the end of World War II, as pressure for production slackened and more time was available for propellant research, experiments with interior-burning grains were begun at CIT. At the time of this writing, the continuation of these experiments at NOTS, Inyokern, indicates great promise for this design for high-performance motors where somewhat longer burning times are permissible. British research on interior burning was already well advanced by the end of World War II. The heating effect on such motors[b] is entirely negligible, and aluminum motor tubes are feasible. This change also will make a significant difference in the performance attainable with rockets of a given caliber.

23.2.7 Weldability

It is frequently desirable to employ welding for attaching nozzles, fins, or lugs to motor tubes. With ordinary mild steel, this introduces no difficulty, but in choosing a high-tensile heat-treated steel for a high-performance motor, its weldability must be considered. No research on this point was done by CIT, since the effect of welding on various types of steel is well known to metallurgists. In general, the very high-carbon steels undergo a marked coarsening of the grain structure and become brittle, so that even very small welds cannot be made without preheating the whole tube. For example, with the N–80 oil well casing used for the 11.75-in. AR[c] motor, it was found impossible to weld on even a row of $\frac{7}{16}$-in. studs 9 in. apart, while on the 5.0-in. *high-velocity aircraft rocket* [HVAR] motor, using NE 8735, no difficulty was experienced with the considerable tack-welding required to attach two suspension lugs and four fin lugs. The difference in composition between these two steels is shown in Table 3. It is easy to be overcautious in this regard, since a slight weakening of the motor tube in local spots apparently causes no trouble if the proper steel alloy is used.

TABLE 3. Compositions of steels used in 3.25-in. and 5.0-in. aircraft rocket motors (NE 8735) and in 11.75-in. motor (API N–80 casing).

Element	NE 8735	N–80
Carbon	0.33–0.38	0.40–0.43
Chromium	0.40–0.60	0.08
Manganese	0.75–1.00	1.50
Molybdenum	0.20–0.30	0.16
Nickel	0.40–0.70	0.12
Silicon		0.025
Sulphur		0.040
Copper		0.20

23.2.8 Threads

FAILURES OF V THREADS

With the exception of the target rocket which used a piston ring closure (see Section 18.5.1), all CIT rockets have had threads at the front end of the motor tube for attaching to the head, and many have been threaded also at the rear to take the nozzle. The standard V-shaped thread is not well suited to the requirements of rocket motors, where strong joints between a relatively thin tube and a usually much thicker piece of steel are desired. It has been almost universally used, however, be-

[b] High-impulse motors, using internal-burning grains, were developed in small sizes at the Allegany Ballistics Laboratory [ABL] under Section H, Division 3, NDRC.[6,7] As of October 1946, the Hercules Powder Company, operating ABL for the Bureau of Ordnance, had developed motors using 100-lb internal-burning grains.

[c] The 11.75-in. aircraft rocket, usually called "Tiny Tim" or simply "Tim" for short, was the last and biggest fin-stabilized rocket developed by CIT.

[redacted]

cause of the much easier availability of dies for this shape than for any other and because all machine shops are experienced in cutting it. For a given thread depth, a V thread is weaker against a straight end force than, say, a square thread, but this effect is not important in rocket motors since no cases are known to the writer in which a motor thread has been "stripped" in the ordinary sense. The thread difficulties that have arisen (and they were relatively rare) were caused by the expansion of the tube by internal pressure and aggravated, in the case of a V thread, by the large angle between the

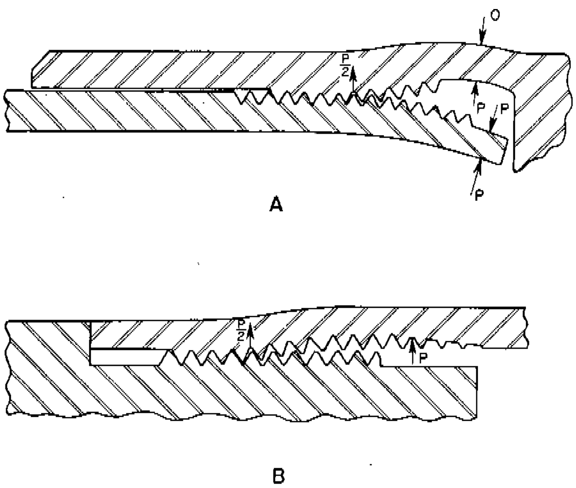

Figure 2. Diagrams illustrating probable mechanism of thread failures on rocket motors at high pressure for (A) 3.25-in. Mk 7 motor, (B) 5.0-in. Mk 1 motor. Arrows denote approximate pressures acting on various surfaces, where P is the total motor pressure.

loaded faces so that a large component of the end thrust is transferred into radial pressure. With external threads on the motor tube, the pressure tends to make the threads tighter, and, if the piece into which the thin tube screws is relatively thick (as is usually the case), no trouble is experienced. With 3.5-in. heads on the 3.25-in. AR motor,[d] however, where the thicknesses of the two threaded pieces are comparable, heads have in several cases been blown off by abnormally high pressures in high-temperature firing with so little damage to either thread that the pieces could be reassembled.

The probable explanation of this phenomenon is as follows. Since the threads are certainly not pressure-tight except in rare instances and some leakage of gas occurs, it would be expected from the high impedance to gas flow of the interstices between the threads that the pressure would drop approximately uniformly along the length of the thread engagement from full motor pressure at one end to atmospheric pressure at the other. This has been confirmed qualitatively by a static firing experiment with the 5.0-in. HVAR motor. The effect of this pressure gradient is illustrated, on a very exaggerated scale, in Figure 2A. The front end of the motor thread is floating in a region of high pressure and hence is not expanded, whereas the adjacent portion of the head is being expanded by the full motor pressure. This expansion allows the full motor pressure to creep farther along through the thread and further accentuate the effect. The result is that only the threads at the extreme rear are holding the motor and head together and only they will be damaged appreciably when the pieces separate.

Special Thread Shapes

For internal threads on the motor tube, the effect is obviously much worse (see Figure 2B) because both the internal pressure and the large obliquity of the loaded faces tend to expand the motor tube but have no effect on the heavy piece screwed into it. For this reason, consideration was given to other thread shapes for the 5.0-in. motors, but experiments indicated that they would probably not be necessary, and experience has confirmed this fact. Only on the 11.75-in. and 14-in. motors,[e] where no advantages from the practical manufacturing standpoint were realizable with V threads, was a special thread shape adopted. Experience with these two motors has indicated that the buttress thread used on the latter is probably the optimum thread shape both with regard to performance and ease of manufacture. The 7-degree angle of the loaded face is small enough to be almost certainly less than the angle of repose between steel surfaces, so that the end thrust produces no slippage and expansion of the tube, and yet it is large enough to provide adequate tool clearance and allow the use of thread hobs of relatively large diameter.

It is well known that maximum strength is obtained when the depth of the thread is one-third the thickness of the tube. This rule is useful as a

[d] The combination of a 3.5-in. head with the 3.25-in. motors Mk 6 or Mk 7 is designated as 3.5-in. *aircraft rocket* [AR]. See Figure 4 of Chapter 19.

[e] The 14.0-in. aircraft rocket motor was a NOTS project initiated in 1945.

[redacted]

guide, although in most cases it is preferable to use a standard thread even though its depth deviates somewhat from the optimum.

Alignment

Ordinary commercial threads cannot be depended upon to hold parts in accurate alignment because of the relatively large clearances necessary to assure interchangeability. To eliminate this difficulty, it has been CIT's practice to use relatively loose-fitting threads (No. 2)—obviously desirable also from the standpoint of easy assembly under adverse field conditions—and to depend for alignment upon screwing solidly against a shoulder. The gas malalignment (see Section 24.8) sets a lower limit of approximately $^1/_{10}$ degree below which improved alignment of the rocket parts does not improve the performance; hence we have made it a universal policy that any two rocket parts must screw together with a malalignment not exceeding this figure. With reasonable care and proper machining setups, this accuracy is attainable in threading operations without increasing the cost, but the methods of specifying and checking it are difficult to establish. The specification finally adopted as the most satisfactory was that the thread seating faces (i.e., the ends of the tube) must be parallel to each other and perpendicular to the mean axis of the tube within $^1/_{20}$ degree and that a "go" thread gauge with a shoulder must seat against the tube ends with a gap not to exceed 0.001 in. per in. of diameter.[f]

23.2.9 Straightness

The existence of a bow in the tubing has an important bearing on the nozzle alignment and hence must be controlled. On rocket motors of 3.25-in. caliber and smaller, the CIT practice was to bend the completed motor[g] so that the nozzle exit cone axis coincided with the center of a "perfect" head or of the front end threads within $^1/_{20}$ degree. On the 3.25-in. AR motors, the bend was made at or near the front end of the nozzle (which is obviously where it belongs), but shorter motors were bent approximately at the middle for practical reasons.

Motors 5 in. in diameter and bigger were not practicable to bend, so alignment was secured by specifications on the tubes and nozzles and their threads separately. Tubing lengths in which the bow was excessive were straightened prior to machining.

23.2.10 Reaction with Propellant

Because corrosion of steel is very rapid in contact with smokeless powder, it is necessary to prevent the grains from touching the bare motor walls. This has been done by painting the inside of the motor tube either with standard Navy projectile-cavity paint or with clear ethyl cellulose lacquer. The lacquer is probably preferable because it gives a smoother and harder finish.

23.2.11 Spinner Motor Tubes

Only two important factors enter into the design of spinner motor tubes which do not appear with finners. The first is the requirements of the bourrelets. On the three calibers of spinners which were tested by CIT, three types of bourrelet were used. Five-inch spinners had the bourrelets on the tube, which was machined full length on the outside. On the 3.5-in. spinners, the rear bourrelet was the nozzle ring, and the front bourrelet was the rear of the head, which was slightly larger in diameter than the rest of the head. Some experimental 2.25-in. spinners were of uniform diameter over the whole length, having no bourrelets.

For barrage or general purpose spinners, any of these methods is probably satisfactory. The difficulty with uniform diameter rounds is that the tubing is never straight. If the outside of the tube is machined, the stress relief resulting from the machining accentuates the bow in the tubing and makes the use of bourrelets necessary for reasonable accuracy.

In the development of the aircraft spinner, there was some evidence that a very high degree of accuracy of the bourrelets is necessary (maximum ovality not more than 0.002 or 0.003 in.) to obtain minimum

[f] That no standard Navy drawings of CIT rockets contain this specification is a result of the Bureau of Ordnance rule that manufacturing drawings cannot specify gauging methods. The usual statement, that "threads shall align within $^1/_{20}$ degree" is almost meaningless operationally and has caused continual confusion.

[g] The apparatus employed for bending tubes is described in references 8 and 9.

[redacted]

dispersion. Like all questions of accuracy, however, it is difficult to settle, and research into the causes of spinner dispersion did not reach the point where any general rules could be stated regarding permissible tolerances.

The second problem peculiar to spinners is the centrifugal force. Its effect is a peripheral tension in the tube which acts like an internal pressure. The magnitude of this pressure is easily calculated by considering the centrifugal force on unit area of the tube and is given in absolute units by

$$P = \rho r t s^2. \tag{2}$$

If we take the density ρ to be 7.3 g per cu cm, measure the radius r and thickness t in inches, and specify spin velocity s in revolutions per second, this is

$$P = 0.027 r t s^2 \text{ (in psi)}. \tag{3}$$

The fastest spin rocket developed by CIT was the 5.0-in./14 GASR Model 39A, having a maximum spin at 70 F of 309 rps, which gives for the pressure equivalent to the centrifugal force approximately 840 psi. Apparently the effect of centrifugal force on the motor tube becomes important only at extremely high spins, but its effect on powder breakup does cause motor bursts as has already been noted.

23.3 NOZZLES

The functions of a rocket nozzle from the viewpoint of interior ballistics has been discussed in Chapter 21, where it was shown that the important characteristics of a nozzle are its throat diameter, which determines the equilibrium pressure of the motor, and its expansion ratio, which determines the amount of additional thrust which can be wrung out of the gases during their expansion. This additional thrust, expressed quantitatively by the nozzle coefficient C_N is determined in practical cases by the expansion ratio, since the divergent angle of the exit cone is never made so large that its effect is appreciable. Obviously, since the gas in the throat is moving with the velocity of sound, a nozzle with a 45-degree half-angle of divergence would have a very low nozzle coefficient regardless of its expansion ratio, since the gas could not expand rapidly enough to touch the exit cone at any point. Half-angles from 6 to 15 degrees have been used, and little is known of the behavior of more rapidly diverging nozzles, although it is probable that they would give decreasing accuracy as well as decreasing thrust.

The ideal interior contour of a nozzle is determined by the desire for maximum accuracy and minimum erosion. Both these considerations favor very long nozzles with gradually tapering entrances and exits. It has been repeatedly demonstrated that whenever the gas is required to change its direction abruptly, local erosion is severe. On accuracy, the evidence is less clear-cut, but it appears that the exact contour of a nozzle is unimportant provided that (1) it possesses axial symmetry and (2) that the flow of gas delivered to it is uniform.[10] In practice, however, the gas flow to the nozzle is not uniform because of the complicated shapes of grains and grids; hence longer nozzles give better accuracy because they have more time available for straightening out this nonuniform flow.[11,12]

Considerations of space, weight, and ease of fabrication dictate that nozzles are always made short and with simple contours. Thus the exit portions are always conical, and the entrance is a combination of straight lines and circular arcs. It has never been possible to obtain a clear correlation between dispersion and any characteristic of the entrance portion of the nozzle other than its alignment. The varied shapes which exist have resulted from considerations of manufacturing methods, necessity for fitting grids, and esthetics.

23.3.1 Nozzle Types

It is difficult to lay down any very useful general rules for deciding which type of nozzle is preferable for a particular rocket. In CIT's case, the choice tended to be influenced greatly by the type of machine tools that were available to us at the time, since the project was doing both design and production. For fin-stabilized rockets, the basic choice is between single nozzles and multinozzles. Single nozzles are obviously the choice for small motors (2.25-in. and smaller) because they are simpler and cheaper to manufacture and because, with multiple nozzles, each nozzle would be so tiny that its erosion would be large. In the large calibers (5.0-in. and larger) the advantage in ease of manufacture probably lies with the multinozzle and three other advantages become important:

1. The possibility of having a central nozzle with a blowout disk, thus increasing the safety at high

[redacted]

temperatures and greatly extending the usable temperature range;[h]

2. A considerable saving in length and perhaps a slight saving also in weight, although the latter is uncertain since no single-nozzle large motors have been made;

3. A decrease in dispersion resulting from the averaging out of "gas malalignment" between the various nozzles.

In the intermediate sizes (3.25-in.), the choice is difficult. That only single nozzles have been used is an indication not of their superiority but of the fact that the advantages of multinozzles became apparent gradually during the existence of the project. For the 3.25-in. AR motor, for example, the abandonment of the single-nozzle design in favor of multinozzle was recommended to the Bureau of Ordnance by CIT early in 1945, and a thorough investigation would probably reveal that some of the other rockets of this caliber could be improved by the change. The best argument for multinozzles—the blowout disk—is, however, less cogent for low-performance and nonaircraft rockets.

23.3.2 Single Nozzles

The simplest way to make a nozzle is to shape the rear end of the motor tube into the proper contour (see Figure 4A). Such "integral" nozzles have been used extensively by the Army, whose rockets have relatively thicker walls than CIT's, and were used on several early rockets. In some instances, nozzle and tube were made separately and butt-welded instead of being formed from a single piece. On the target rockets they were satisfactory because the accuracy was almost completely controlled by the fins, but on the CWR (see Section 18.4) they were abandoned for accuracy reasons. It was never possible to manufacture them without appreciable variations in thickness around the nozzle throat. Hydrostatic pressure tests [13] and experiments with the yaw machine [12] showed that these variations caused the exit cone of the nozzle to deflect under the pressure of the firing, thus changing the axis of thrust. If a really good fabrication method were available, integral nozzles would have important advantages in saving weight and eliminating several manufacturing operations, but it seems clear that the ordinary methods of swaging and spinning cannot make nozzles of sufficient accuracy, at least in the range of wall thickness which has been investigated.

The inaccuracy obtained with integral formed nozzles is largely eliminated if both ends of the nozzle are held firmly by a piece of tubing so that the exit cone cannot deflect appreciably. Hence separate formed nozzles [i] inserted into the motor tube have been used successfully on the majority of CIT rockets. They have been made rapidly and cheaply by a number of techniques [14,15] with sufficient accuracy to be acceptable, although again the chief difficulty with them is accuracy.

Nozzles machined from bar stock were used, on the *antisubmarine rocket* [ASR] and *barrage rocket* [BR], for example, before acceptable techniques for forming nozzles had been developed. Functionally they are preferable to any other, since they can be made as accurately as desired, but they cannot compete in mass production with the formed nozzle except in small sizes where screw machines are readily available. Thus in CIT production the formed nozzle for the 2.25-in. SCAR [j] cost 75 cents to make and 25 cents to braze into the tube. The nozzle for the BR was very similar, but, machined from bar stock, it cost more than twice as much.

Attachment of Single Nozzles

For attaching machined nozzles to motor tubes, two methods have been used in quantity production, as shown in Figure 3. The use of threads is probably not ideal because of the objection to internal threads on the motor tube discussed under *Special Thread Shapes*, in Section 23.2.8, and because it is difficult to be certain that the threaded joint is moisture-tight. Unless care is taken to tighten the nozzle firmly, it may move slightly when the pressure comes on the tube, thus introducing a malalignment. Nevertheless, threaded nozzles were used extensively on low-performance motors and were satisfactory. The specification of thread alignment with the seating face naturally applies to the nozzle threads as well as to the tube. Threads cannot be

[h] Several gadgets for achieving the same result with a single nozzle were tried but showed little promise.

[i] In early CIT reports this type of nozzle is often called a re-entrant nozzle.

[j] The latest official designation of the forward-firing practice rounds is "2.25-in. Forward-Firing Aircraft Rockets (Target)." In most of the literature they are known as *subcaliber aircraft rockets* [SCAR]. Three variations are distinguished by Mark numbers. See Figure 7 of Chapter 19.

[redacted]

used with formed nozzles because the thin wall will not accommodate the necessary seating shoulder.

For 1.25-in. motors, the swaging method shown in Figure 3B is probably the best solution. The joint is rigid when properly made and is well adapted to quantity production.

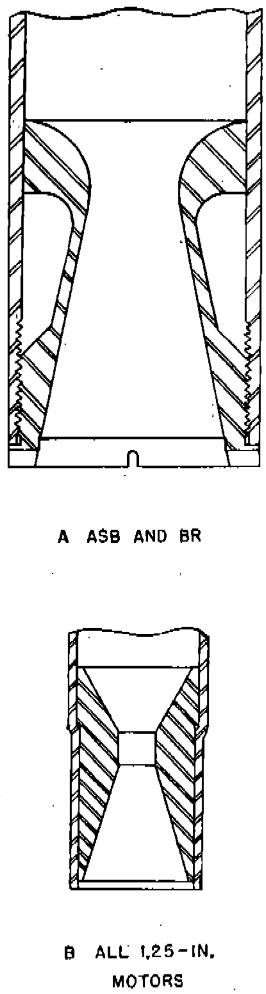

FIGURE 3. Methods of attaching machined nozzles.

Copper brazing or silver soldering has been used for most formed nozzles and for a few machined nozzles. A smooth joint is formed and, particularly with induction heating, the rate of production is good, and the damage to the motor tube by the heat is negligible because the critical part of the tube (just ahead of the nozzle) does not get very hot. With the relatively thin formed nozzles it is desirable that both ends fit snugly into the motor tube since otherwise one runs into the same warpage difficulty as with the integral formed nozzle, although on a reduced scale since here it would be the entrance of the nozzle rather than its exit cone which would be shifted by the warpage. To avoid having to press the nozzle in for its full length, a procedure which usually results in galling the inside of the tube and rolling up metal ahead of the nozzle so that the grid does not seat properly, three alternatives have been used (see Figure 4).[k]

1. On the 2.25-in. SCAR, the rear end of the tube was machined internally for the length of the nozzle to a diameter nominally equal to the nozzle external diameter, so that clearance or interference up to 0.004 in. was possible in the most adverse cases. The fact that the front end of the nozzle could have a few thousandths of an inch freedom was accepted in the interest of easier production, since the nozzles were relatively thick in proportion to their diameters and the accuracy of a practice round was not of prime importance. Heavy press fits were eliminated by selective assembly when necessary.

2. On the 3.25-in. AR Motor Mk 7, a bead was rolled or pressed into the tube so that the nozzle would drop into the tube loosely from the rear and be tight for the last one-quarter inch approximately. In order to meet the two requirements that the rear end be a press fit and that there be a small clearance for the silver solder, a 0.002-in. step was machined on the rear contacting surface of the nozzle. This method of attachment was evolved after considerable experience with others and is believed to be the best.

3. On the VAR series[l] (3.25-in. Motor Mk 1 et al.), the nozzles were made as shown in Figure 4D because leaving the tube with its full 3.25-in. diameter at the rear allowed so little airflow through the 7.2-in. tail that the stability of the rockets would have been unduly low. The same design was adopted for the first AR motor (3.25-in. Mk 6) in the interest of standardization but soon abandoned because it has little to recommend it. The complicated shape was much more difficult to make than the bead in the Mk 7 motor, and the reduction of the tube diameter ahead of the nozzle was undesir-

[k] Several other possibilities were tried on the BR but were abandoned because of increased dispersion. They are illustrated in reference 16.

[l] The series now designated "7.2-in. retro rockets," designed for firing backward from aircraft, has more frequently been called *vertical antisubmarine rockets* [VAR]. Velocities of 175, 200, 210, 310, and 400 fps are obtained with 3.25-in. motors of different length but identical design. See Figure 2 of Chapter 19.

[redacted]

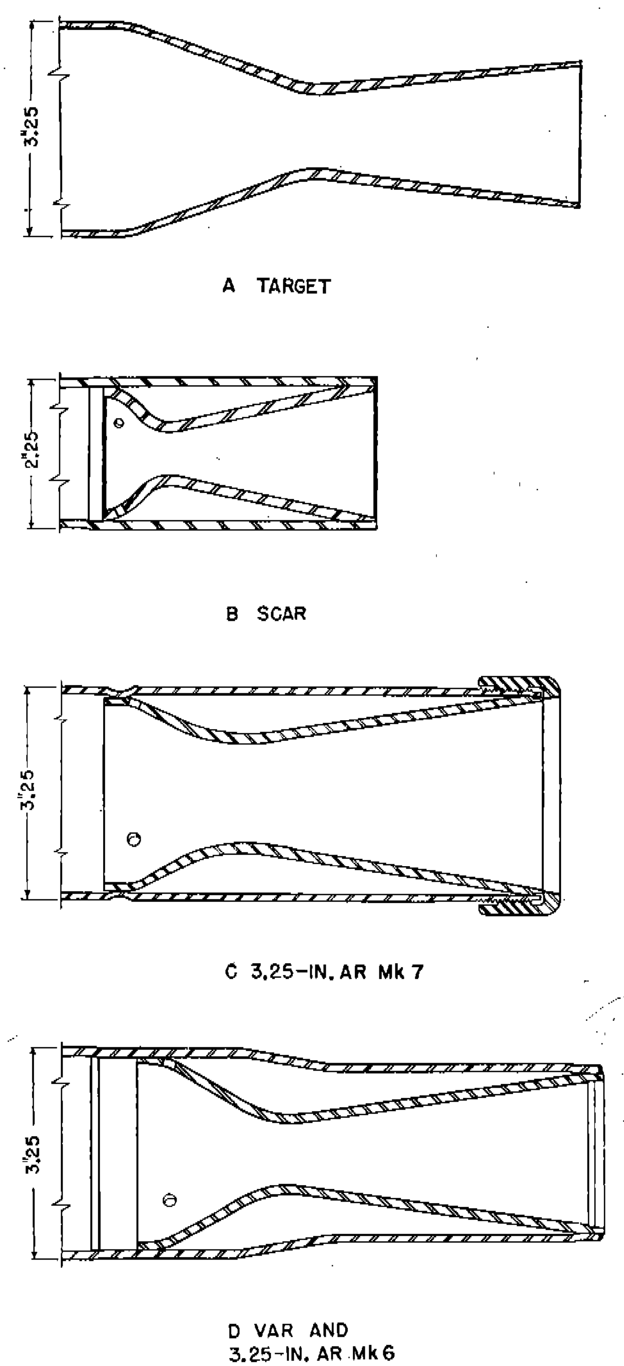

FIGURE 4. Methods of attaching formed nozzles.

able because it increased the internal K of the motor. On low-performance motors the change in internal K was not critical, but on the aircraft rocket motor it gave an easily measurable reduction in the upper temperature limit. A further disadvantage, which again is most significant for high-performance motors because of their large nozzle throat, is the reduction in nozzle expansion ratio entailed by the swaging down of the tube.

Because the silver solder joints were usually the weakest point of the motor, it was standard practice to give them a thrust test with a force corresponding to the product of the internal cross-sectional area of the tube by the maximum expected motor pressure with an appropriate safety factor. A considerably stronger joint can be made by arc welding, as was done on the VAR's and some others, but this technique is not favored because it leaves a rough exit circle. In addition to the obvious objection of the necessity for cleaning up the weld, the roughness has a more subtle fault. Since the gas is discharged from the nozzle at a pressure above atmospheric, it exerts a radial pressure on the nozzle exit cone, and, if the cone is slightly longer on one side than the other, there will be a net side force which is small in magnitude but large in effect because of its very long lever arm relative to the center of mass. Tests with the 3.25-in. AR[17] indicated that the deflection so introduced was of the order of 2 mils per 0.01 sq in. of unbalanced area.

Considerably thinner stock can be used for forming nozzles if the motor pressure is given access to the annular space between the nozzle and the tube. On the British RP–3, the annular space is sealed from the inside of the motor by an obturator cup because it is open to the outside through the fin slots. The CIT practice, on the other hand, has been to provide ports between the annular space and the inside. If this is not done, the combination of the pressure gradient between the motor and the annular space and the setback force of the grain will collapse a thin nozzle at the throat. The holes are placed so that any lubricating oil or cleaning compound which might be trapped in the annular space will drain out when the motor is stood on the front end, since otherwise it might seep out after the rocket is loaded and react with the propellant.

23.3.3 Multiple Nozzles

The first problem facing the designer of a multinozzle rocket is the number of nozzles to use. For fin-stabilized rounds, where malalignment is important, the choice is considerably narrowed by the rule that the nozzle arrangement should have the same symmetry as the grain so that the amount of gas flowing through different nozzles is equal or at least symmetrically arranged. For example, a motor containing a cruciform grain should have four or eight nozzles or a multiple thereof, whereas six nozzles are appropriate for a triform grain. That this rule is necessary is based on good logic and poor experimental evidence, but it has been followed because it turned out to be convenient to do so. The experimental evidence consists of (1) a firing of six rounds of four-nozzle CWR's with three-ridge grains which flew wildly for reasons unknown [18] and (2) the fact that the nozzles of the 5.0-in. spinners which are shielded by the legs of the grain do show less erosion in static firing than those opposite the openings.

Consideration of nozzle erosion is important, since its effect is much greater on many small nozzles than on a few larger nozzles having the same total throat area because of the greater exposed surface of the smaller nozzles. The change in total nozzle area is roughly inversely proportional to the nozzle radius, so that, unless it is possible to adjust the progressiveness of the grain to compensate for the increased nozzle area (as was done on the cruciform charges for 5.0-in. spinners), one will not get good burning curves if the nozzle radius is too small.

With these two factors in mind, one usually chooses the number of nozzles primarily on the basis of the space available in the nozzle plate. There is probably an optimum number from the viewpoint of manufacturing cost, since the lower unit cost of making a small hole is balanced by the larger number of them required, but this is not a very critical criterion.

Multinozzles can either be machined directly in a nozzle plate or made individually and inserted into a relatively thin plate. The former "integral" type has been used in finners and the insert type in spinners because of the disparity between the amounts of propellant in the two types. A glance at the nozzle plate of an HVAR or a Tim will show that so much of the area is taken up with nozzle that if one is to have an adequate expansion ratio (approximately 4 is usually considered desirable), there would be almost no metal between nozzles of the insert type, and the plate would not withstand the motor pressure. If one were to make a low-performance finner with a propellant charge comparable to those which, because of the length limitation, are used in spinners, he might choose the insert-type nozzle plate. It has been used exclusively on spinners primarily because of its considerably smaller weight—approximately 4 lb for the 5.0-in. spinner compared to 7.5 lb for the HVAR.

In the matter of cost, the advantage lies with the insert nozzles because a slip in machining one nozzle hole does not result in scrapping the whole assembly. Thus in CIT production of over 100,000 motors, the one-piece nozzle plate (with its skirt or ring) for the HVAR cost $11.87. Despite its much greater complexity, the nozzle assembly for an eight-nozzle spinner could have been made for less than $8.50.

The individual insert nozzles have been made as simple as possible with a cylindrical outer surface in order to keep the cost down. CIT purchased 5.0-in. spinner nozzles at 10.3 cents each. Putting a shoulder on them to keep them from being blown out by the motor pressure requires a considerable increase in machining cost. Copper brazing was universally used for holding the nozzles in the plate, although silver solder would be equally good, and other suggested methods (such as pinning) appear to have no functional disadvantage provided that the nozzles are not loose in their holes.

23.3.4 Nozzle Tolerances

Since nozzles are difficult to manufacture because of their complicated shape and this difficulty increases greatly as the specifications and tolerances are made more stringent, it would be extremely useful to be able to define precisely the limits within which inaccuracies in fabrication will not noticeably affect performance. This is never even approximately possible in practice because in any borderline case it is the dispersion that is in question, and dispersion is extremely difficult to measure precisely. It is influenced by such a diversity of factors difficult to control that, unless the factor being considered has a very large effect (as is seldom the case), one can seldom say with certainty whether the difference in dispersion between two sets of field firings was the result of the factor in question or not. It is, of course, also true that no borderline between good and bad nozzles exists, but all gradations between best and worst appear. In setting standards of acceptance for nozzles, one is thus continually required to make arbitrary decisions with little or no assistance from the experimental facts. A few general principles are available to guide the decision, and these are listed in the following

[redacted]

paragraphs. But beyond these, the best that can be done is to assume that the ideal nozzle is perfectly smooth and perfectly symmetrical in all details and to reject on principle any manufacturing method which gives nozzles differing more from the ideal than those made by another method. Thus hot spinning was abandoned by CIT when other forming techniques became available which gave smoother interior surfaces, even though the effect of smoothness was not very firmly established experimentally.

The throat diameter of a single-nozzle motor affects only the operating pressure, but its dimension is not very critical because the variation in surface area among different grains is usually about ± 1 per cent. A variation of the same amount in nozzle throat area corresponds to such a large variation in diameter that tighter tolerances have been specified on the drawings in order not to encourage sloppy workmanship. On multinozzle motors, uniformity of nozzle diameter is required to keep down the malalignment.

The thickness of a nozzle must be great enough at every point to withstand the setback force of the grain (and also the pressure differential in case the nozzle is not vented), but the uniformity of thickness is important to guarantee that it does not distort unsymmetrically when the pressure and heat are applied and thus introduce dispersion.

On machined nozzles it is not feasible to blend the entrance and exit cones into a smooth curve, and a short cylindrical surface is left at the throat. The length of this flat does not appear to influence dispersion if it is small compared to the throat diameter, but sharp angular transitions between it and the conical portions have been avoided lest there be a tendency for the gas to pull away from the surface. The latter consideration may not be significant because a sharp angle would erode away very quickly.

The surface smoothness is unimportant within rather wide limits. Certainly nothing is to be gained by honing or polishing the interior of a nozzle to a better finish than that of ordinary cold-rolled steel (about 100 microinches)[19a] and a considerably rougher finish would probably be satisfactory except for the fact that it has not been possible to devise a gauge for checking the direction of the axis of a rough nozzle. Nothing can be learned by firing rough nozzles, since the direction of their alignment is not accurately known. Gouges or ridges or other imperfections are to be avoided if they are unsymmetrical around the periphery, especially if they are in the throat or exit cone. The entrance cone appears to have no effect on accuracy unless it is displaced or cocked at a considerable angle with respect to the throat and exit cone. The effective axis of the nozzle is almost exclusively determined by the axis of the throat and exit cone.

Ovality of the throat or exit cone is undesirable for the same reason as roughness—the alignment-checking mandrel will not determine the actual effective axis of the nozzle, and, if this uncertainty is much greater than $^1/_{10}$ degree on the average, an increase in dispersion will result.

For multiple-nozzle plates on finners, we have the additional requirement that the average alignment of the nozzles must be perpendicular to the thread seating face within the usual $^1/_{20}$ degree, since on such large motors it is not practicable to bend the tube to bring the nozzle axis into coincidence with the center of mass. The alignment of any particular nozzle can be allowed to vary by several times this amount. A similar requirement is necessary for spinners, although here the tolerance depends on the stability factor. That the effect is significant in practice despite the averaging of the malalignment by the rotation was shown by a test on the 5.0-in. HCSR Model 134,[m] in which cocking the nozzle plate ½ degree deflected the rocket 14 mils from its trajectory for zero malalignment. To guard against such a consistent error in cant angle for several nozzles in one plate, a fairly close tolerance on cant angle was specified.

23.3.5 Flash Suppression

The elimination of the luminosity of the rocket jet is desirable in some applications for concealment and is particularly important for forward-firing aircraft rockets, where the flash may temporarily blind the pilot during night combat. It was found that single-nozzle rockets having small nozzle expansion ratios gave very luminous trails during the whole of burning, the brightness being greater at higher temperatures. A nozzle with a large expansion ratio apparently cools the gas below the flash point before allowing it to mix with the air, so that the trail is invisible except for an instant at ignition and again when the grain collapses at the end of burning.

[m] In the standard CIT designation, HC denotes the head type (high-capacity) and SR denotes spin-stabilized rocket.

With the divergence angle usually used (6 to 9 degrees half-angle), the minimum expansion ratio for flashless performance at all temperatures is close to 4.0, but it appears to be larger for nozzles with much larger divergence angles.[19,20]

For multinozzle rockets, the situation is probably little different, although no comprehensive investigation of the effect of expansion ratio on flash has been made. The HVAR, having an expansion of 4.0, was rendered flashless by adding a "nozzle skirt," a piece of 5-in. tubing which extended 1 in. behind the rear face of the nozzle plate. A multinozzle with an expansion ratio of 5 will give flashless performance as a nonrotating rocket but not as a spinner.[21] No way is known to eliminate the flash of spinners.

23.3.6 Erosion

We have already seen that the rate of heat transfer per unit area from the propellant gas to the rocket metal parts is almost proportional to the "weight velocity" and hence is largest at the rear end of the grain and in the nozzle throat. The decrease in tensile strength with high temperature does not ordinarily cause collapse of the nozzle because the throat is of small diameter and has a relatively thick wall, but it does cause erosion which increases the nozzle area during burning at a rate dependent primarily on

1. The material of which the nozzle is made.
2. The temperature of the propellant gas.
3. The motor pressure.
4. The burning time.
5. The shape and size of the nozzle.

Despite the large discrepancy of approximately 2700 F between the melting point of steel and the temperature of the gas, it does not appear that any appreciable melting occurs, since, long before the melting point is reached, the strength of the steel becomes insufficient to withstand the high stresses imposed by the flowing gas, and plastic flow occurs in the metal.[22] That this explanation of erosion is correct is indicated by the photomicrographs of Figure 5 which show a typical case of relatively severe erosion such as is encountered with the small-diameter spinner nozzles shown in Figure 6. Figure 5 shows a section near the nozzle throat. The grain structure of the metal in zone A, next to the surface, shows that it has been heated above the Ac_3 point

FIGURE 5. Panorama of inside area of cold-rolled steel nozzle. Portion of region near nozzle throat shows the changes in grain structure due to the temperature gradient. Zone A shows structure corresponding to Ac_3 (1560 F). Zone B corresponds to Ac_1 (1365 F). Zone C did not reach the Ac_1 temperature.

[redacted]

(1560 F); that in zone B indicates heating above the Ac_1 point (1365 F) but below the Ac_3 point; and that in zone C has not been altered, so that the temperature must have remained below the Ac_1 point. The

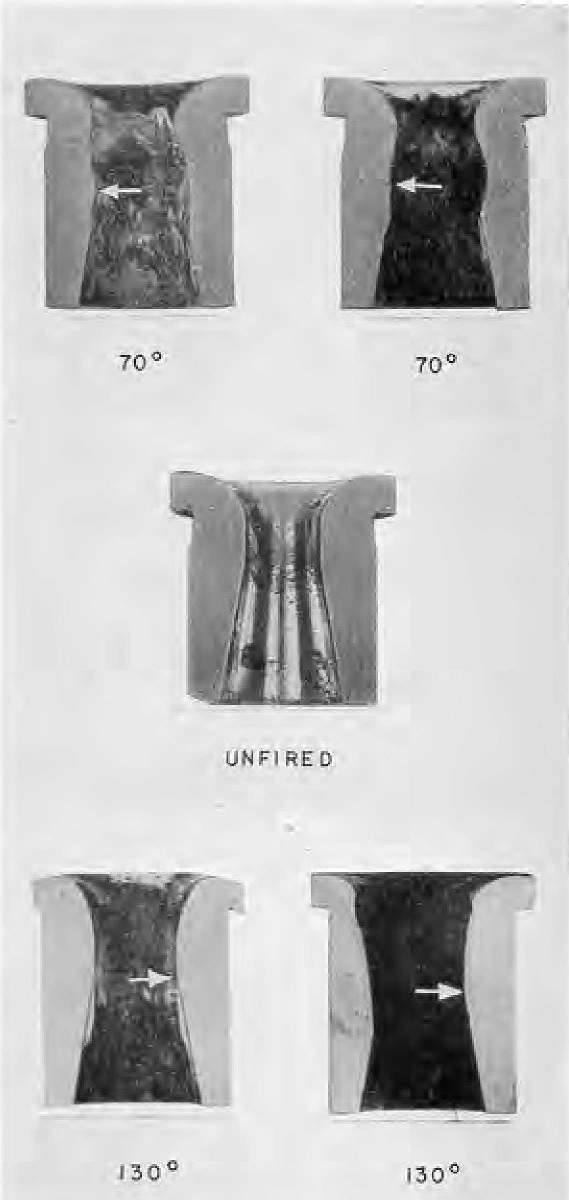

FIGURE 6. Typical erosion of small steel nozzles.

maximum temperature reached in zone A cannot be determined metallurgically, but extrapolation would indicate a value in the neighborhood of 1800 F. The photomicrograph of a section from the same nozzle in the exit cone (Figure 7) shows the unaffected steel which had not been heated to the Ac_1 point, and a superimposed layer of steel which had flowed from the throat, having been above the Ac_3 point but not melted. The temperature distribution in the nozzle is plotted in Figure 8. It should be noted that the temperature is high only in a very thin surface layer.

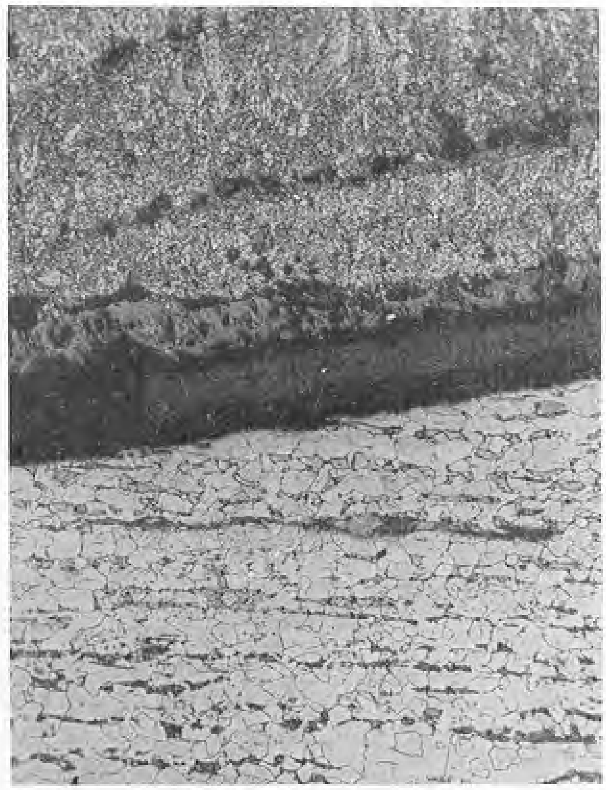

FIGURE 7. Photomicrograph (132 x) of section of exit cone of steel nozzle. Lower part is cold-rolled steel nozzle proper which has not reached the Ac_1 point. Upper part is metal which has plastically flowed from nozzle throat.

This analysis of nozzle erosion shows that the melting point of a nozzle material is of importance only indirectly in that the tensile strength tends to be low near the melting point. For a heat-resistant nozzle, the important factor is high tensile strength in the neighborhood of 2000 F. Two types of heat-resisting materials have been suggested—the ceramics and the high-melting-point metals. No ceramic that has been tried even approaches the requisite strength. Even with a powder weight of only 1.5 lb as used in the ASR, the ceramics cracked and eroded so severely as to reduce the final pressure to less than half its normal value, whereas ordinary steel

[redacted]

nozzles on this rocket can be fired several times. Molybdenum and tungsten nozzles show virtually no erosion at all. Tungsten carbide, which can easily be cast into the proper shape, also works well, but because of its brittleness it must be properly supported. Thus a nozzle throat insert of tungsten

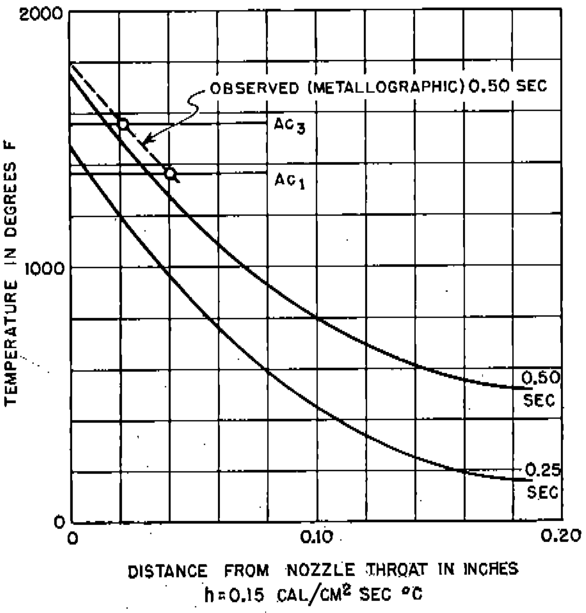

FIGURE 8. Temperature distribution in steel nozzles; $h = 0.15$ cal/cm²·sec·°C.

carbide with a cylindrical outer surface cracked severely when fired, whereas with a conical surface no cracking occurred. None of these heat-resisting nozzles have been used because there was not sufficient need for them to justify the extra cost.

In contrast to the heat-resisting nozzles, the type which has actually been used may be called the heat-absorbing type, which depends upon its ability to cool the surface by conducting heat away from it faster than the gas can supply it. The most important property of such a nozzle material is its thermal conductivity. Thus under conditions where the inner surface of a cold-rolled steel nozzle would reach a temperature of 2040 F and erode away considerably because its tensile strength becomes effectively zero about 400 degrees below this, a copper nozzle would not get above 950 F and would show very little erosion since it would still have some strength at that temperature. The theory of heat-absorbing nozzles is discussed in reference 23, from which Table 4 is taken. Experimental data in the last column of this table are taken from tests at 130 F with the insert nozzles of the 3.5-in. spinner where erosion is especially severe because of the small throat diameter (0.289 in.). The results of these experiments were in complete agreement with the theory. Thus various types of high-speed tool steel were all found to be inferior to cold-rolled steel because their low conductivity more than counterbalanced their greater strength. In particular, Stellite and Hastelloy, special alloys which maintain a high tensile strength even at red heat, gave the highest erosion of all the metals tested, the surface temperature apparently actually reaching the melting point. In Figure 9 is shown a comparison of the Stellite nozzle with one of chromium-plated copper. The latter works very satisfactorily because the copper has an extremely high conductivity (nearly ten times that of steel) while the chromium, although

TABLE 4. Characteristics of nozzle materials.

Metal or alloy	Thermal conductivity at 1600 F (cal/cm·sec·°C)	Density (g/cm³)	Specific heat (cal/g·°C)	Thermal capacity (cal/cm³·°C)	Tensile strength (psi)	Predicted surface temperature $h = .22$ $\theta = .45$ (°F)	Melting point (°F)	Quality as nozzle material	Percentage erosion in actual testing
Hastelloy	0.03	8.94	0.092	0.92	....	2700	2350	Very poor	65
Stellite	0.035	8.38	0.10	0.84	....	2650	2370	Very poor	59
Inconel	0.036	8.51	0.109	0.93	....	2700	2540	Very poor	46
Stainless steel	0.039	8.0	....	....	....	2650	2700	Very poor	....
Monel K	0.062	8.5	0.127	1.06	20,000 at 1600 F	2200	2400	Very poor	....
Cr steel	0.07	7.74	0.11	0.85	0 at 1800 F	2100	2700	Poor	45
Cold-rolled steel	0.0875	7.8	0.168 at 1600 F	1.31	0 at 1600 F	2040	2600	Poor	40
Tantalum	0.130	16.6	0.036	0.60	....	1750	5162	Excellent	....
Iron	0.19	7.8	0.162 at 1800 F	1.26	0 at 1800 F	1750	2795	Fair	....
Molybdenum	0.346	10.2	0.075	0.78	....	1250	4748	Excellent	none
Beryllium	0.385	1.8	0.505	0.94	....	1240	2462	Good	....
Chromium	0.65	6.9	0.187	1.29	....	1070	2939	Good	....
Aluminum	0.66	2.7	0.277	0.75	0 at 600 F	1050	1218	Very poor	....
Copper	0.858	8.9	0.126	1.12	0 at 1000 F	950	1981	Fair	....
Silver	0.97	10.5	0.076	0.80	0 at 1100 F	900	1760	Fair	....

having a somewhat lower conductivity than copper, contributes its high melting point and hardness.

Among the various low-carbon free-machining steels which one would naturally choose for machining a complicated piece like an integral multinozzle, there is little difference in erosion characteristics, but any of them is significantly better than SAE 1020, apparently because of higher manganese content.[24,25] A few sintered and cast nozzles which have been tried have all eroded seriously. No really comprehensive study of nozzle materials was attempted by the project because, at the short burning times in use, the problem was not of sufficient urgency to warrant it. Such surveys have been made by groups interested in liquid-fuel rockets and jet engines.[26]

Despite the shorter burning time, erosion is greatest at high temperatures because of two effects: (1) the higher weight velocity increases the coefficient of heat transfer from the gas to the nozzle wall, and (2) the higher motor pressures cause plastic flow to occur at lower nozzle temperature. The variation of erosion with powder temperature for the case of the 3.5-in. spinner[27] is shown in Figure 10.

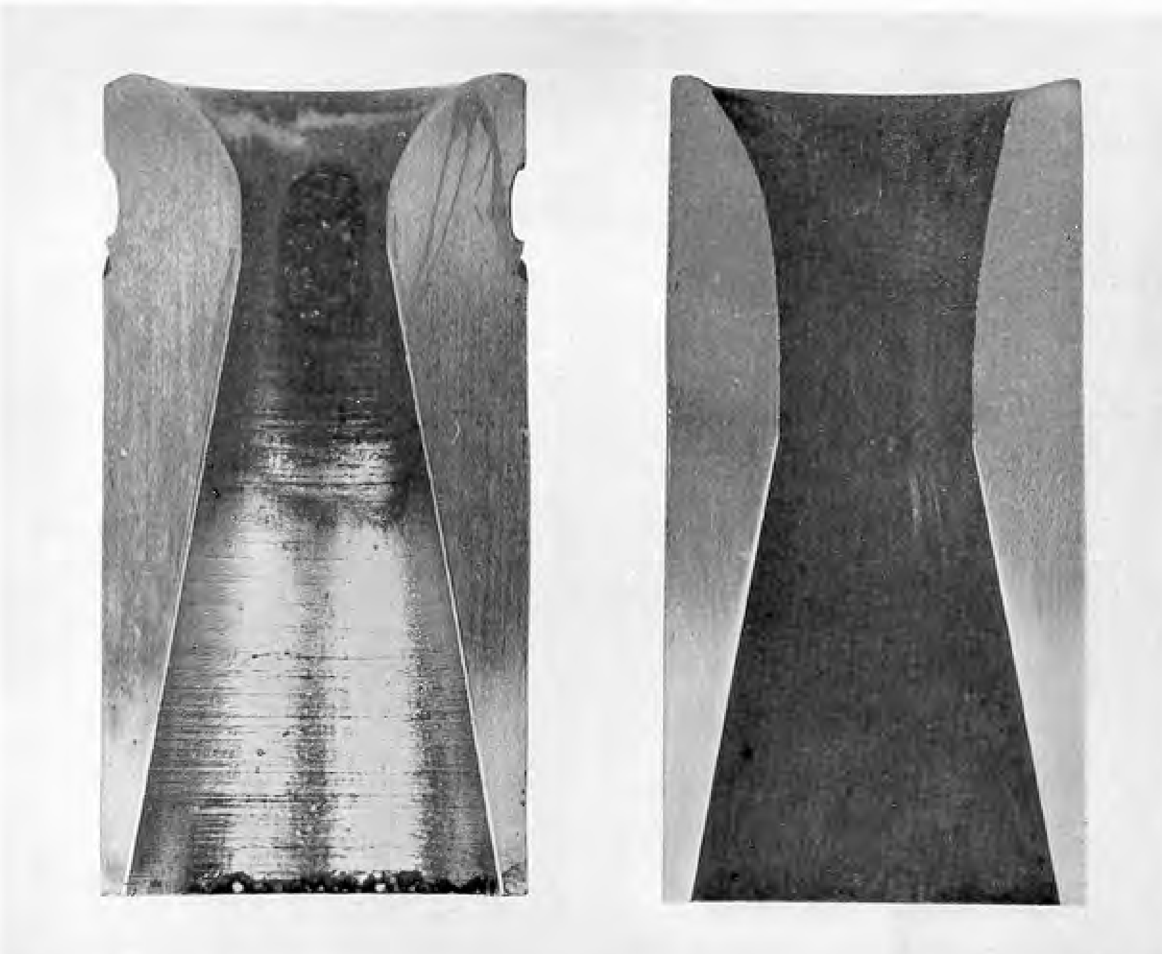

FIGURE 9. Extremes of good and bad nozzle erosion under identical conditions. *Left:* chromium-plated copper. *Right:* Stellite. Initially the nozzles had identical inside contours.

Since erosion is most severe at points where the gas is forced to change its direction rapidly, some improvement can sometimes be obtained by careful attention to the contours. Thus longer entrance cones reduce nozzle throat erosion, but this fact is of little importance because in practice one prefers to have entrance cones as short as permissible.

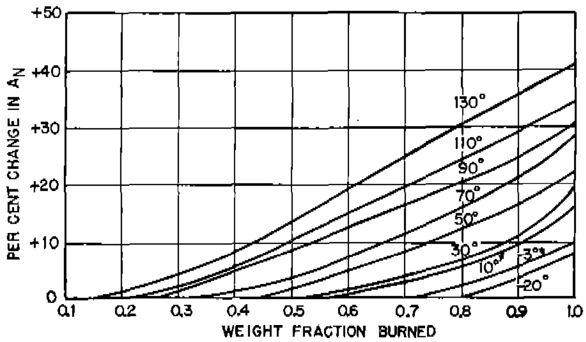

FIGURE 10. Increase in area of small steel nozzles from erosion during burning at various propellant temperatures.

23.3.7

Blowout Disks

For large rockets or for those which are used in situations where a motor burst would involve exceptional hazards (e.g., aircraft rockets), it is desirable to include an extra nozzle in the center of the plate and close it with a blowout disk which is ejected if the motor pressure exceeds a particular value. This device is made necessary by the relatively small strength and large temperature coefficient of the present powder, and as rocket propellants are improved, its use will become less necessary. It allows one to combine the characteristics of two different rockets in one jacket. With the blowout disk closing the central nozzle, the nozzle K is high, say 210 to 220 for ballistite, so that the motor operates at relatively high pressure and short burning time, having its range of useful temperature displaced below that usually designed into rockets. With the central nozzle open, the situation is reversed so that high-temperature performance is increased at the expense of low-temperature performance. If increasing the useful temperature range were the only consideration, the blowout disk would be designed to be ejected at approximately the temperature midway between the two extremes desired. In practice this has not been done because in the vicinity of the blowout pressure it is impossible to predict whether the disk on a particular rocket will blow out or not, the temperature range of this uncertainty being close to 20 F. Since in forward firing the sight setting is influenced very markedly by the burning time, this would mean that in this 20-degree range one would not know what sight setting to use and, if the wrong guess were made, the rocket would be too inaccurate to be useful. On the 5.0-in. and 11.75-in. aircraft rocket motors, therefore, 110 F was chosen as the temperature at which half the disks blow out, this temperature being above that normally required in practice and having a short burning time so that the error in gravity drop caused by a disk's blowing unexpectedly would not be so great as at lower temperatures. Thus the blowout disk has been used primarily as a safety valve rather than as a temperature range extender. Its effect on the temperature range is striking, nevertheless, particularly on the 5.0-in. HVAR, which operates with a very small percentage of failures at 140 degrees. Its lower temperature limit is not known, but it has been fired successfully after having been packed in solid carbon dioxide (sublimation point −109 F) over night.

Blowout disks have been made of annealed copper because a copper disk is thicker for a given blowout pressure than a steel one, and hence small variations in thickness have less effect on the blowout pressure. A disk fails by first bulging out into a hemispherical shape and then shearing. Empirically it has been found that the failure pressure cold (i.e., in a testing machine) can be calculated fairly accurately if a shear strength of 25,000 to 26,000 psi is used. The mean blowout pressure measured in static firing is only 5 or 10 per cent higher than that.

Since rockets with blowout disks are usually designed with relatively large nozzle K's, they will be just as unsafe at high temperatures if the disk should fail to blow as if too small a nozzle had been used. The easiest ways to make an error in this regard appear to be the substitution of too thick a disk or the inclusion of two disks. Because of the importance of having the proper disk, it has seemed desirable to eliminate any possibility of error by two provisions:

1. Each disk is gauged for thickness and its thickness marked on it with a rubber stamp in a position where it is visible from the outside of an assembled motor and can be checked by the final inspectors and the loading crew.

2. The "disks" are made cup-shaped rather than flat so that the ring or grid stool which holds them in place in the nozzle plate cannot be properly assembled if two disks have been used. If two rockets were made with blowout disks of different thickness, it would be desirable to make them completely noninterchangeable by a similar trick.

Proper insulation of the disk from the motor gases is obviously essential to its proper function.

[redacted]

Asbestos and fiberboard have been used successfully, but the best insulation is probably a molded disk of asbestos-filled bakelite. The insulation should fit snugly but not tightly in its hole and be completely covered with a fireproof hard-setting plastic material (like Permatex No. 2) to eliminate any gas leakage around the edges. If the insulation for a blowout disk of small diameter is a press fit in its hole, it may resist being ejected so that an erratic increase in blowout pressure results.

23.4 TAILS

23.4.1 Types of Tails

Fins for rockets show an extreme diversity of size and type. Those on the Army 4.5-in. rocket, for example, are 4 in. long and ¾ in. wide, whereas those on the CIT target rocket measure 18 by 36 in. The target rocket is, of course, a special case, since its fins were made as large as possible for visibility, whereas ordinarily we wish to make fins as small as is consistent with adequate stability. Target rocket fins are, therefore, treated as a separate problem in Chapter 18, and in this section it will be assumed that the primary function of a fin is to stabilize the rocket in flight.

The design of a tail[a] is always a compromise between a number of mutually contradictory requirements. Thus accuracy requires large tails, whereas space and weight considerations and air drag favor small. Simplicity and cheapness of manufacture favor fins made from a single thickness of metal and welded to the tube, whereas weight and convenience may favor double fins with relatively complicated attaching mechanisms. The usual factors controlling fin design include:

1. Adaptability to the type of launching contemplated.
2. Accuracy.
3. Strength.
4. Air drag.
5. Shipping space.
6. Sometimes provision of electrical contact.

Two principal types of tails have been used: ring tails and fin tails.

[a] For brevity we shall use the term "tail" instead of "fin assembly" in referring to the aggregate of the fins, rings, or members of whatever shape which constitute the stabilizing member.

23.4.2 Ring Tails

Ring tails have been used on most rockets having heads of larger caliber than their motors because, except in the unusual case of the ASR where the center of mass of the round is in the head, it is necessary to have a ring behind the center of mass with the same diameter as the head in order to fit a simple launcher. To provide easy electrical contact, the tail has always consisted of two rings, one attached to the motor tube and the other insulated from it. This combination makes the ring tail

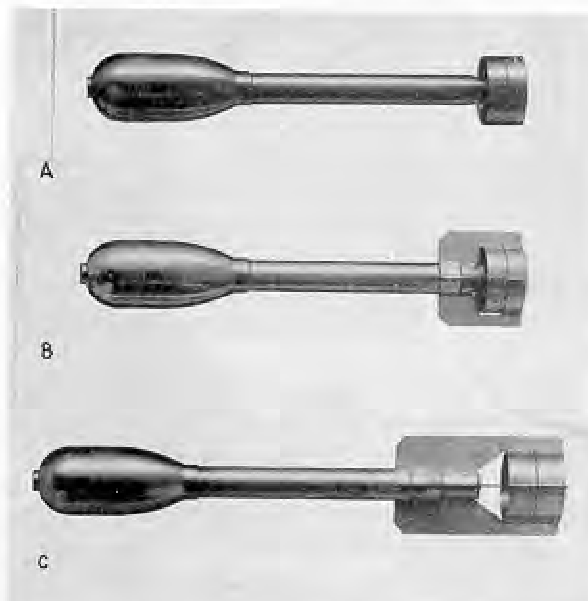

FIGURE 11. CWR tails: (A) plain ring tail, (B) ring tail with radial fins (CWR-N), (C) experimental ring-and-fin tail designed for maximum stability.

rockets adaptable to very rapid loading in any orientation such as is required in combat and to automatic launching. The design is inherently strong, and the thickness of metal used is determined by the rough treatment experienced in handling and in being jammed against the knife edges of the electrical contacts, rather than by the relatively small aerodynamic forces encountered in flight. For withstanding water entry and for stabilizing underwater trajectories, ring tails are very satisfactory.

As used on CIT rockets, the ring tails are not very efficient in stabilizing the rounds because the ring diameters are relatively small and not much air passes through them, especially at high velocities, because of the shielding by the head. Usually the accuracy of the rockets was adequate for the tactical situation, but in the case of the CWR, radial fins extending beyond the ring were finally added, as shown in Figures 11A and B, to increase the stability and decrease dispersion. High-speed water tunnel tests indicated that this change would reduce the yaw oscillation distance σ from 236 to 192 ft, whereas field firings gave values of 260 and 215 ft. The water tunnel tests showed also that a further decrease of σ to 166 ft could be made by moving the tail back about 1 caliber as shown in Figure 11C and that considerably more water passed through the ring under these conditions. Whether the same would be true when the effects of the rocket jet are added is problematical, but it may be possible to increase the efficiency of ring tails by thus moving them back and still retain the advantage that no part of the tail projects beyond the head diameter.

23.4.3 Fin Tails

Fin tails have been used on all the aircraft rockets, and again the choice was dictated by the launching method. Since the motor and head were of the same diameter,[o] fin tails allowed the rockets to be attached closer to the airplane with a consequently smaller drag. The width of the fins (i.e., in the radial direction) was also determined by the space limitations, and in all cases the length was made approximately 1½ times greater than the width in order to obtain the requisite strength. This ratio of length to width appears to be a good one, at least for subsonic rockets.[p]

For rockets small enough to be handled manually, the forces encountered in handling are again the determining factor in the strength. In sizes comparable to Tim, however, it ceases to be practicable to make fins so strong that they will support the weight of the rocket, and the aerodynamic forces are determining. These forces are difficult to calculate, but are not large in practice because appreciable yaws are obtained only at low velocity.

Economy in shipping space demands that the fins be detachable from the motor. In practice this means that fins for large rockets which cannot be boxed in groups of four with the fins nested between them (as was done with the 3.25-in. AR motor) will have fins individually detachable, and hence, to accommodate the locking mechanism, the fins may be made of two pieces of metal, dished so as to leave a space between. The double fins also have the advantage of being relatively strong with thin metal. On the British RP–3, detachability was achieved with remarkable simplicity and effectiveness with a single-thickness fin. Although it is almost certainly the best rocket fin in existence, it could not be copied in CIT rockets because slots in the motor tubes were not permissible.

Drag was mentioned as a factor in fin design, but nothing more has been said about it, and in fact little consideration was given to it in the design of CIT rockets. The reason is that for short-range firing, such as aircraft forward firing, there is little to be gained by small reductions in the drag, since the total drag is large but its effect is small. For long-range rockets, this would not be true.

Another thing about fin design may be conspicuous by its absence—namely, any mention of folding fins. These have been used effectively on the Army 4.5-in. rocket, but were not tried by CIT. The reason is simply that, since integral formed nozzles cannot be used with the thin-walled rockets, there is no place to put the Army type which opens back. Fins which open out sideways are subject to serious objections: (1) inaccuracy, since the first moment after launching is the time when stability is most needed, and (2) practical difficulties in making a foolproof latch to hold them in the open position.

23.5 SUSPENSION LUGS

Suspension lugs and lug bands have been used only on forward- and backward-firing aircraft rockets where the air drag of the launcher is a prime consideration. In almost any other conceivable application, the drag of the rocket itself would be of greater importance, and lugs would be omitted. The drag of the front lug, in particular, can be quite significant because it is placed in a portion of the rocket which would otherwise usually be aerodynamically "clean," and its presence thus increases the turbulence along almost the full length of the rocket.[28]

The shape of the lugs being dictated by the shape of the rocket and the method of launching, little

[o] The 5.0-in. AR with the 3.25-in. motor is an exception to this rule, but there the controlling factor was the use of a motor already in production.

[p] See discussion under HVAR in Chapter 19.

[redacted]

can be said about it in general. It should be noted, however, that the shape now standard on the 3.25-in. and 5.0-in. aircraft rocket motors is certainly not ideal, resulting as it did from the historical accident that long T-slot launchers were already in combat use before the advantages of post launchers (or "zero-length" launchers) were established. The front lug is not very strong, is difficult to manufacture, and has more drag than would be desired. The ideal would probably be two lugs side by side about 90 degrees apart on the front and one between them on the rear, so that the rear lug could be made higher and stronger than is now the case and still not interfere with the front post when it passes. This type of suspension was used for experimental aircraft firings of Tiny Tim from fixed wing launchers.

Whenever their use is possible, welded lugs are much preferable to lug bands because (1) they assure that the position and spacing is always correct, (2) they are easier to make strong enough, and (3) they are cheaper to manufacture.

23.5.1 Strength of Lugs

The basic data for determining the required strength of a lug is usually in the form of the maximum values of yaw, roll, and pitch which the aircraft is expected to undergo in the most extreme maneuvers contemplated or possible. The translation of these specifications into forces in various directions on the lugs is obvious and straightforward and typically results in strength specifications which are difficult to meet. Fortunately, the basic data by their very nature contain a considerable safety factor, so no additional factor need be interjected. For carrier-based planes, there is also a specification of the maximum fore-and-aft accelerations encountered by the airplane in catapulting and arrested landings, but the maximum fore-and-aft force which the rocket itself will experience usually depends upon vibration, in the case of wing-mounted rockets, and its magnitude is difficult to estimate.

Most of the difficulties with lug bands are eliminated or greatly reduced if the bands can be made tight enough. In the case of the 11.75-in. motor, "tight enough" meant going to specially heat-treated high-tensile steel. The best design for the tightening mechanism on a lug band is probably that shown in Figure 12, which was adopted for the 5.0-in. and 11.75-in. motors after experience with several other types. In case slippage along the tube is undesirable, as it is for the post launcher where only one post contains a latch, it can be eliminated by drilling a shallow flat-bottom hole in the motor tube and having a pin on the lug band which projects into it. This was done on the nonwelded CIT design of the HVAR (5MA4). (See Section 19.4.2.) The drill marks on the 3.25-in. Mk 7 motor tube served to position the lug bands when they were attached but did not significantly reduce the slippage because of their tapered sides.

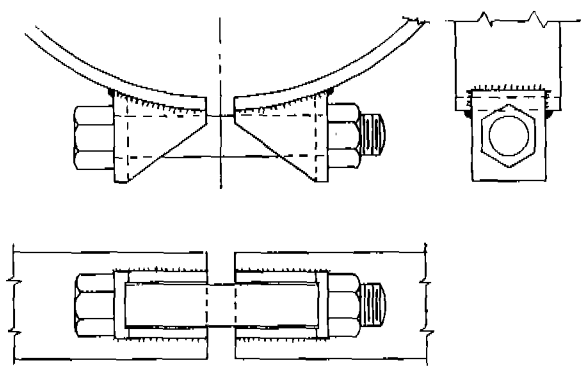

FIGURE 12. Final design of lug band clamp.

23.6 MOTOR SEALS AND GRAIN SUPPORTS

Two factors in motor design have not yet been mentioned. That of making electrical contact to the igniter is a rather simple and specific problem. It is mentioned in Chapters 18 and 19 in connection with the 4.5-in. BR, the 3.25-in. AR, and the 11.75-in. AR motors, but it will not be discussed here in detail.

The problem of supporting the grain at the front end and sealing the two ends of the rocket against moisture are, on the other hand, practically identical for all rockets, except for scale effects.

23.6.1 Grain Support

In order to eliminate the possibility of cracking the propellant grain as a result of impact against the grid when the igniter fires, it has been considered desirable to hold the grain at the extreme rear end of the motor by means of some type of

[redacted]

grain support at the front end. For small-diameter motors, where the weight of the grain is small and its length short, the front motor seal is adequate for this purpose. It is simply pushed in until it seats firmly against the igniter which in turn contacts the grain. When the length of the grain exceeds about 2 ft or its weight becomes of the order of 10 lb, this simple procedure is not adequate. If the grains are not thoroughly annealed, they shrink with age as the strains introduced during extrusion are relieved. Temperature changes also cause changes in length which can be significant on very long grains, and it is desirable to have something to take up these length changes without allowing the grain either to become loose or to exert so much force on the front sealing disk that the seal is broken. The best substance which has been found for doing this is a thick felt disk compressed to about $\frac{2}{3}$ or $\frac{3}{4}$ of its unconfined length. Felt has no undesirable effects on the propellant, nor is it affected by the propellant fumes. Felt disks are used in both the 3.25-in. and 5.0-in. aircraft rocket motors.

With the heavier grains, the accelerations experienced during handling might be large enough that the grain would move the front seal if it were not reinforced. In the 3.25-in. and 5.0-in. motors, this support is provided by the front thread protector. The 11.75-in. motor is special in that the grains are held against the grid by the charge support independently of the motor tube.

It is interesting that the only test which was made of the necessity for holding grains firmly against their grids showed that it was not necessary. Two rounds of the 5.0-in. HVAR were fired at 120 F with 20-lb heads which would give them an acceleration of more than 80g. The rounds flew normally, although the grains, with grids attached, were separated from the grid stools by distances of $3\frac{1}{4}$ and $4\frac{5}{8}$ in.[17] Despite this evidence, the requirement that grains be firmly seated is based on good logic, particularly since in some cases the grid can rotate if it becomes loose, a circumstance which would almost certainly cause a motor burst.

23.6.2 Seals

The first seals used by CIT to keep moisture out of motors were binderboard and fiberboard disks pressed into position and sealed with glyptal lacquer. When tests had demonstrated that such disks did not in fact keep out moisture if the motors were subjected to extreme temperature changes, cellulose acetate was substituted, and it in turn was found to be inadequate and displaced by steel. The complete story of the tests made to determine the best seals is contained in reference 29, and will therefore be merely summarized here.

No nonmetallic materials were found which, either in the shape of disks or cups, would protect motors from extreme conditions of exposure. Fiberboard absorbs water through even the best paint seals, swelling up and softening. Thermoplastics have a thermal coefficient of expansion much greater than that of steel, and as a result plastic closures shrink away from the motor tubes when

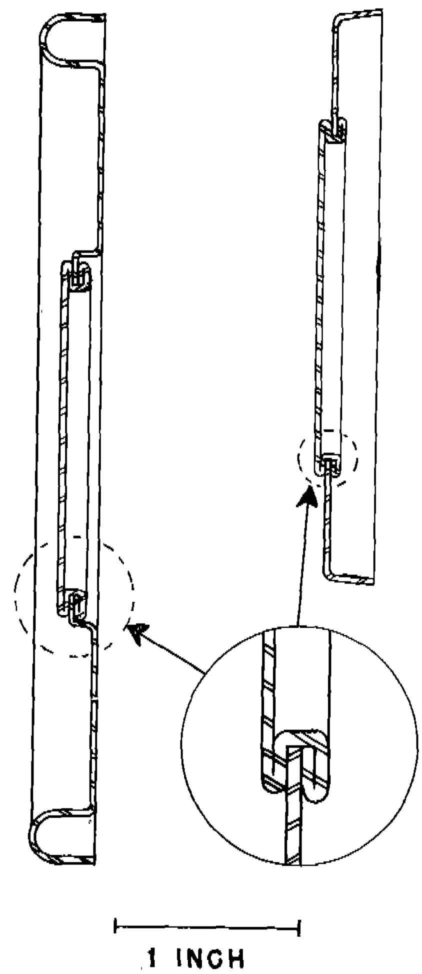

FIGURE 13. Front end motor seals for 5.0- and 3.5-in. motors.

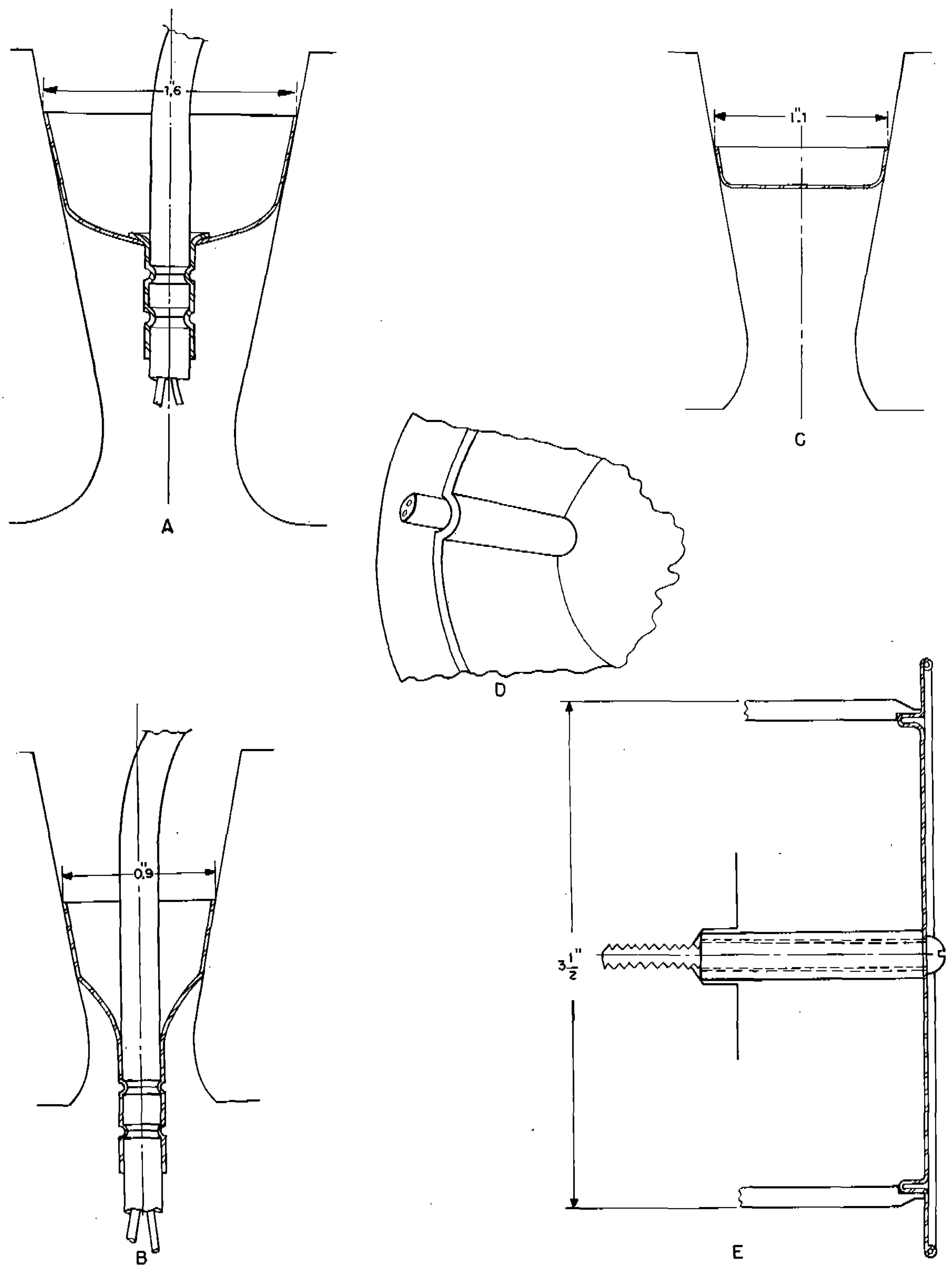

FIGURE 14. Metal nozzle seals: (A) SCAR; (B) HVAR pigtail seal (for one nozzle); (C) HVAR plain seal (for other 7 nozzles); (D) BR, CWR, and similar motors with electrical contacts on tail use seal similar to (C) but with two wires brought around seal edges at opposite sides as shown; (E) 3.5-in. spinner.

[redacted]

they are cooled, breaking the paint seal at the edges. Bakelite is too brittle to be inserted tightly without cracking.

Of all metal closures, steel seems to be unquestionably the best. Brass closures corrode rapidly in contact with steel motor tubes because of electrochemical action. Aluminum corrodes even more rapidly, large holes being eaten away leaving the covering film of glyptal unsupported, and in addition it is too soft so that it was difficult to insert aluminum closures without deforming them. Steel closures are easy to construct, can be inserted rapidly, and, if they have the proper thickness so that they still have some spring in them after being inserted, they give effective protection against moisture even without a perfect paint seal. They have the same coefficient of expansion as the motor tubing and the nozzles. Some objections have been raised to them because of their missile hazard, but tests indicate that it is no more serious than with fiberboard disks.

Three types of seals are required for a rocket motor. For the front motor seal, the most effective design appears to be a flat-bottomed cup either with plain or re-entrant sides as shown in Figure 13. The "blowout patch" in the center was evolved after considerable experimentation as the best device for opening quickly at low pressures but still being easily moisture-proofed when it is in place. In some motors it serves the purpose of admitting the gas to the pressure-arming base fuze, and in all motors it assures that the motor would not become propulsive in case of accidental ignition when the head was not screwed on.

For nozzle seals, a simple shallow cup with tapered sides is adequate when no wires must come through it. To accommodate the igniter leads, the cup must be made slightly more complicated as shown in the examples in Figure 14. Even when good nozzle seals are used, care must be taken that moisture does not enter through the nozzle threads (if any) or along the cotton insulation or filler in the igniter leads.

For the 3.5-in. and 5.0-in. spinner motors, it appeared simpler to seal the nozzle end with a single metal disk instead of closing each nozzle separately. Thus the wires connecting to the contact rings are completely enclosed and protected. The standard nozzle end seal for the 5.0-in. motors is shown in Figure 4 of Chapter 20. The seal for 3.5-in. motors, shown in Figure 14E of Chapter 23, is basically similar but has a flange extending beyond the diameter of the round to keep it from sliding forward in tubular launchers.

For sealing all these steel cups, the best material found is General Electric glyptal red lacquer No. 1201, with the addition of 7 per cent by weight of aluminum powder, which toughens it and makes it dry better around wires insulated with nylon. Nozzle closures hold better if the edges are painted with thinned glyptal containing emery, 200-mesh being the optimum granulation.

The larger motors are so expensive that extra precautions have seemed desirable to keep them dry, and auxiliary seals have been used at both ends. At the front, the extra seal is easily incorporated in the thread protector, but the design of the rear one depends on the motor. Blowout patches may be required in these also to keep the motor from being propulsive when shipped.

[redacted]

Chapter 24

EXTERIOR BALLISTICS OF FIN-STABILIZED ROCKETS

By *C. W. Snyder*

24.1 INTRODUCTION

In the following two chapters, we shall discuss briefly and qualitatively the exterior ballistics of rockets. It is an exceedingly large field and one of considerable complexity; we shall attempt merely to lay a groundwork for understanding why rockets behave in flight as they do and what methods are used to predict their performance, and to indicate where more thorough discussions of various aspects of the problem can be found. The theoretical basis of the subject is treated in detail in a book to which we shall refer frequently by the abridged title of *Exterior Ballistics*;[1] this is the source of much of the following material.

24.1.1 Specification of a Rocket's Motion

It will be well to have clearly in mind at the outset the precise meanings of the terms which will be used in the description of the sometimes complex motions of a rocket in flight and the symbols by which they will be denoted.[a] There is an important theorem of mechanics which states that the motion of the center of mass of a solid body which is acted upon by any arbitrary combination of forces is the same as if all the body's mass were concentrated at that point and all the forces acted on that point. Consequently, the simplest way to treat the motion of a solid body and the way that is always adopted in practice is to consider first the motion of the center of mass and then independently of this motion to consider the rotations of the body about the center of mass.

The path of the center of mass through space is called the *trajectory* of the rocket. It is in general a complicated curve, but the simplest case, when it lies in a vertical plane, is illustrated in Figure 1. When the rocket (represented by a small arrow in Figure 1) is at the point C, its center of mass is moving in the direction of the tangent to the trajectory (shown as a dashed line intersecting the horizontal coordinate axis). The angle θ between the initial orientation of the rocket (i.e., the launcher orientation) and the tangent to the trajectory at a particular time we shall for brevity call the *trajectory deviation*. In addition to θ, we must of course know the orientation of the plane of the trajectory (i.e., the plane containing the launcher line and the tangent to the trajectory) in order completely to specify the rocket's direction of motion. The angle θ is the more important quantity, however, because,

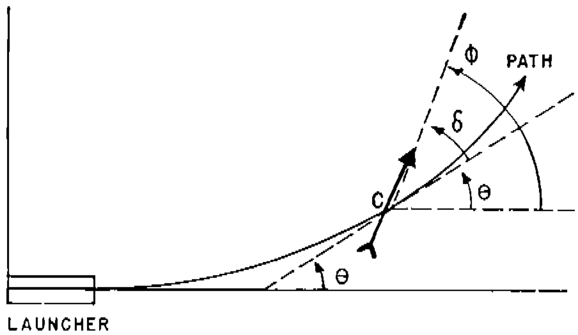

FIGURE 1. Trajectory of rocket in vertical plane.

except for gravity, nearly all the forces acting on a rocket are unchanged when the orientation of the plane is changed.

In general, the rocket will not be pointed in exactly the direction in which its center of mass is moving, and in this case it is said to have a *yaw*. The rocket can yaw in any direction,[b] but for finners the usual case is that shown in Figure 1, where the yaw is in the plane of the trajectory. The *yaw angle* δ is the angle between the trajectory and the axis of the rocket.

A third angle which is usually of less importance than either θ or δ is the *rocket orientation angle* ϕ. It is the angle between the rocket axis at any time and the line through the launcher. In the plane case shown in Figure 1 we have obviously the relation:

$$\phi = \delta + \theta,$$

[a] The notation here is taken from *Exterior Ballistics*[1] and agrees in the main with that of earlier CIT reports.

[b] This usage of the term *yaw* is slightly different from the nautical usage. Thus a ship *yaws* sideways but *pitches* up and down.

but this will not hold in general unless we consider the angles as vectors, a complication which we will avoid here.

24.2 FORCE SYSTEM OF A FINNER

Because of its complex shape, both interior and exterior, a rocket is subject during flight to a multiplicity of complicated forces, and an understanding of rocket motion requires that we replace this force system with a simpler one which produces the same accelerations and velocities. An elementary theorem of mechanics assures us that it is always possible to do so. The resulting force system is, of course, arbitrary, and it is chosen to make the analytic representation as simple as possible. In particular, it is convenient to consider separately the forces arising from the combustion of the propellant and those arising from the presence of the atmosphere, since the former disappear after the end of burning.

24.2.1 The Jet Force and Torque

From consideration of the conservation of linear momentum, we derived in Chapter 21 the fact that the ejection of the propellant gas from the nozzle results in a force on the rocket which was called the thrust. For simplicity we shall assume that its direction and magnitude are constant throughout burning and that it ceases abruptly. Actually, of course, its time variation is given approximately by the pressure-time curve (see Chapter 21), but the assumption of constancy introduces fairly small errors, which are discussed in *Exterior Ballistics*.[1]

In the ideal case, the line of action of the resultant jet force would lie along the rocket's long axis and pass through its center of mass. Since these conditions are never perfectly fulfilled, we obtain, in addition to the forward thrust, a torque of magnitude equal to the product of the thrust by the distance between its line of action and the center of mass. This is the so-called "jet malalignment torque."[c]

One other subtle torque results from the action of the gas on the rocket. If the rocket is rotating about a transverse axis during burning, the gas as it flows down the motor tube will have to be accelerated laterally. The reaction on the motor tube tends to damp the rotation. This so-called "jet damping torque" is too small to be important in practice.

[c] Actually there may be two types of malalignment,[1a] but only one is important in practice.

24.2.2 Aerodynamic Forces

The effect of air on the rocket in flight can be treated with sufficient accuracy by means of two forces and two moments, defined as follows. Consider a projectile moving through still air in the direction of the vector **V** of Figure 2, having a yaw represented by the angle δ and having a certain instantaneous angular velocity about the transverse axis perpendicular to the plane of the paper through its center of mass. Although the aerodynamic forces are produced by the distribution of pressure over the entire surface of the projectile, we need not consider the distribution in detail because its effect is the same as that of a suitable single force F_A acting at an arbitrarily chosen point (for convenience taken to be the center of mass) plus a suitably

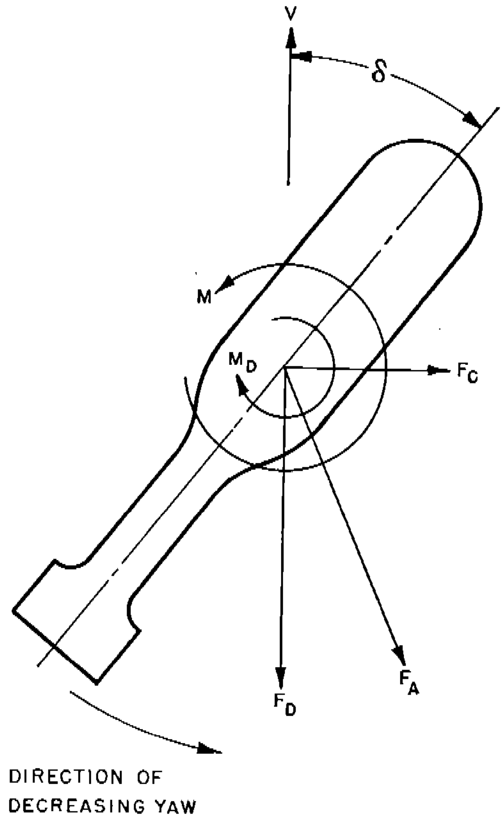

FIGURE 2. Aerodynamic forces and torques acting on fin-stabilized rocket.

[redacted]

chosen torque. If F_A is resolved into components along and perpendicular to the trajectory (i.e., to the velocity vector **V**), the former is the "drag" F_D and the latter is the "cross-wind force" or "lift" F_C. Of the total torque, the major part, which depends upon the yaw but not on the transverse angular velocity, is called the "righting moment" or "restoring moment" M since it tends to reduce the yaw; and the small part which varies with the transverse angular velocity is called the "damping moment" M_D because it tends to reduce the angular velocity and momentum. In the figure it is assumed that the yaw is decreasing so that M_D, tending to oppose the decrease, acts in the direction opposite to M. When the yaw is increasing, both moments tend to oppose the increase. The principal effect of the drag is to decrease the velocity and range of the rocket, whereas that of the righting moment is to stabilize the rocket and to produce oscillations in the orientation of the rocket whenever it yaws. The cross force and the damping moment are of relatively minor importance and serve chiefly to damp the oscillations. It was noted in Chapter 21 that a righting moment exists for small yaws only if the fins are sufficiently large, and that F_D, F_C, and M are nearly proportional to the square of the velocity V, up to about 800 fps. Hence we set

$$F_D = mV^2c; \tag{1}$$

$$M = \mu V^2 \sin \delta \approx \mu V^2 \delta; \tag{2}$$

where c is the deceleration coefficient, m the mass, and μ the righting moment coefficient. Equations (1) and (2) are equivalent respectively to equations (16) and (22) in Chapter 21.

The force system of Figure 2 is not, of course, the only one that will produce the same acceleration of the rocket as does the actual pressure distribution. It is possible to find a point on the axis of the rocket such that the resultant force F_A applied at this point gives the moment M and hence is fully equivalent to the entire aerodynamic pressure distribution. This point is called the center of pressure, and the force system is shown in Figure 3. It is more convenient than that of Figure 2 for visualizing the effect of aerodynamic forces but less useful for computation. The center of pressure must lie to the rear of the center of mass in order for a finner to be stable.

If the rocket is traveling through water or earth, the aerodynamic forces are replaced by a different force system, which will be discussed later.

24.2.3 Other Forces

To complete the list of forces which determine a rocket's trajectory, the pull of gravity and the reaction of the launcher must be included. The latter is effective for such a short time that it can be considered as an impulse.

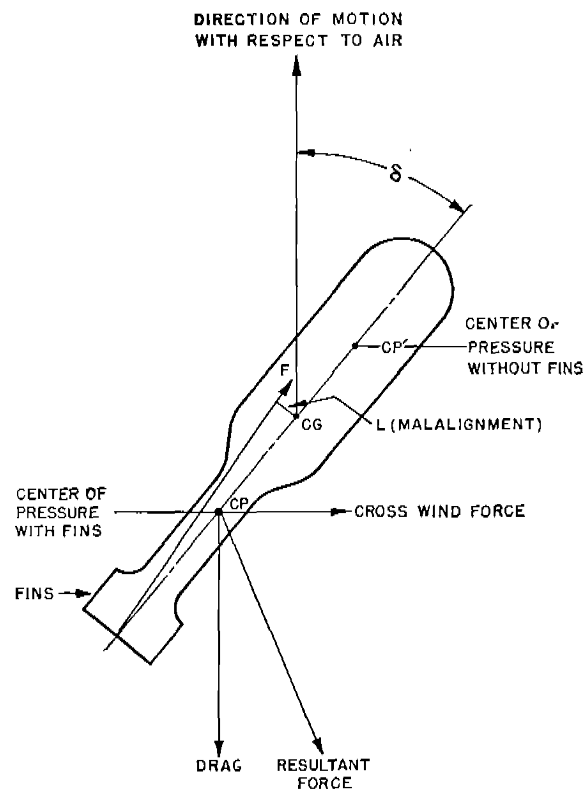

FIGURE 3. Alternative aerodynamic force system.

24.3 USE OF THE FORCE SYSTEM

Through an analytical representation of these various forces and torques, it is possible to set up the equations of motion of a rocket in flight, both during and after burning, and, at least theoretically, to solve them for the motion of the rocket under various initial conditions. This analysis is developed in detail in *Exterior Ballistics*.[1] In practice, of course, the solution of the equations is extremely difficult unless a number of simplifying assumptions are made.

[redacted]

24.4 RANGE OF A GROUND-FIRED ROCKET

The quantity of first concern is usually the range of the rocket or, more accurately, the mean range of a large number of identical rockets fired under the same conditions. For this calculation, one assumes that the thrust, the drag, and the pull of gravity are the only forces acting. We shall see that the solution of even this much simplified case is very difficult unless the velocity is small.

The vacuum range X of a projectile in free flight after launching at velocity V_0 and elevation angle θ_0 was given in Chapter 21 as

$$X = \frac{V_0^2 \sin 2\theta_0}{g}. \tag{3}$$

It was noted that this requires modification for rockets on two counts: (1) it must be corrected for the burning time, since the rocket is not in free flight until after the jet force ceases, and (2) the effect of aerodynamic forces cannot in general be neglected. Even as slow and dense a rocket as the "Mousetrap" antisubmarine rocket [ASR] (175 fps) attains only 95 per cent of its maximum vacuum range.

The effect of burning time is most conveniently allowed for by the concept of the "equivalent shell." [d] An equivalence is set up between the rocket and a hypothetical shell which have coincident trajectories after the rocket stops burning. Thus, having translated the initial conditions of projection of the rocket into those of the equivalent shell, we can use equation (3) or other more exact relations from shell theory to determine the range and trajectory of the rocket subsequent to burning. The expressions for accomplishing this translation are as follows.

If a rocket is fired at a quadrant elevation greater than zero, its velocity at the end of burning will fall short of that calculated from momentum considerations [equation (6) of Chapter 21] for two reasons: there is a component of gravity acting backwards along the trajectory and the air resistance is continuously removing energy during the acceleration. The actual "burnt velocity" will be, instead of V_0 to be expected in a vacuum,

$$V_b = V_0 - t_b\,(g \sin \theta_0 + \tfrac{1}{3} c V_b^2), \tag{4}$$

in which t_b is the burning time (duration of thrust). The factor ⅓ takes care of the fact that we should actually use some kind of average velocity during the burning period instead of V_b itself in computing the effect of air resistance. These same two effects will reduce the velocity of the equivalent shell, but they will have only half as long a time to act, since the rocket, starting from zero velocity, has during the burning time an average velocity half that of the shell. Hence the shell, if it is to match the rocket flight in space and time, will be fired later than the rocket by half the burning time and will have an initial velocity

$$\begin{aligned} V_e &= V_0 - \tfrac{1}{2} t_b\,(g \sin \theta_0 + \tfrac{1}{3} c V_b^2) \\ &= V_b + \tfrac{1}{2} t_b\,(g \sin \theta_0 + \tfrac{1}{3} c V_b^2). \end{aligned} \tag{5}$$

Finally, because of the greater gravity drop of the rocket, the equivalent shell will be fired at a lower angle of elevation than the rocket by an amount proportional to the length of time which the rocket burns beyond the launcher. In fact, the elevation angle of the equivalent shell will be

$$\theta_e = \theta_0 - \frac{1}{2}\frac{g}{V_b}(t_b - t_p) \cos \theta_0 \text{ (in radians)}, \tag{6}$$

where t_p is the time spent on the launcher (designated by the subscript p because at the time this theory was developed, the term "projector" was in vogue).

24.4.1 Air Drag

It is immediately evident that no ballistic calculations can be made without a knowledge of the value of the drag of the rocket at all velocities that it attains. Aerodynamics has not progressed to the point where the drag coefficient of a projectile can be computed on purely theoretical grounds, but by a combination of theory and empirical results it is possible to make surprisingly close estimates of the drag coefficient of an aerodynamically clean projectile. However, if it has large lugs, fin braces, or other irregular projecting parts that tend to produce large contributions to the drag, the estimation is much more difficult. Examples of such calculations are given in *Exterior Ballistics* [1] and in references 4 and 5.

The method employed is to divide the total drag into five parts:

1. Head resistance.
2. Base drag due to reduced pressure at the rear.

[d] The theory of the equivalent shell is worked out in references 2 and 3.

[redacted]

3. Skin friction of the cylindrical motor tube.
4. Skin friction of the fins.
5. Drag due to lugs and other irregularities.

Parts 1 and 2 can be estimated from the known drag of a shell having a nose shape as much as possible like that of the rocket under consideration. The skin frictions, 3 and 4, are calculable from aerodynamic theory. When the contribution of 5 is not significant, the sum of parts 1 to 4 is often in fairly good agreement with the experimentally determined drag. An interesting and important example in which this is not the case is analyzed in Table 1.

TABLE 1. Relative contributions to total drag for 3.5-in. aircraft rocket (CIT Model 5).

Velocity (fps)	Head resistance and base drag	Motor skin friction	Fin skin friction	Unaccounted for	$\frac{C(V)}{C(600)}$ (experimental)
600	32%	18%	34%	16%	1.00
800	33%	17%	33%	17%	1.00
1,000	39%	15%	30%	16%	1.06
1,200	37%	8%	15%	40%	2.05
1,400	$46\frac{1}{2}$%	$7\frac{1}{2}$%	$14\frac{1}{2}$%	$31\frac{1}{2}$%	2.07

Columns 2, 3, and 4 give the theoretical estimates for various parts of the drag, expressed as percentages of the total experimentally determined drag. Column 5 gives the percentage of the total drag which the theoretical analysis does not account for. Presumably most of this discrepancy is caused by the unusually large lug bands which the motor carries in order to accommodate 5.0-in. as well as 3.5-in. heads. Column 6 gives relative values of the total deceleration coefficient at various velocities. Theoretically the increase in drag between low and high velocities should be approximately 3 to 2 rather than 2 to 1 as actually observed, illustrating the well-known fact that good streamlining is much more important above sonic velocity.

24.4.2 Calculation of Range

If one has precise knowledge of the value of the deceleration coefficient as a function of velocity, it is theoretically possible to make accurate trajectory and range calculations by means of numerical integration, but the labor involved makes such calculations impracticable except on a modern mechanical or electrical integrator, few of which are in existence at present. Complete ballistic tables have been worked out for several different shell shapes, and what is done in practice is to pick the one of these standard drag functions which most nearly approaches that of the rocket in question and use the ballistic tables corresponding to that function.

This method of calculation is quite satisfactory for low-velocity rockets, i.e., in the velocity range where the deceleration coefficient can be assumed constant. For firings at quadrant angles below 15 degrees or for segments of a trajectory in which the direction of the trajectory does not change by more than about 30 degrees, the Didion-Bernoulli method [1b, 6] is probably the most convenient. For rockets fired from the ground at higher quadrant elevations, the Otto-Lardillon tables [7] have been reduced to more convenient graphical form in reference 2 and have been found to be sufficiently accurate and very useful for predicting ranges.

The greatly increased complexity of the problem at higher velocities arises from the varied shapes of rockets. Skin friction, the turbulent drag of the projections, and the other factors will contribute to the total drag in varying proportions for various rockets, and each factor will, in addition, vary with velocity in a different manner. Thus no one drag function can be expected to be a sufficiently good approximation for more than a very restricted family of rockets. Several resistance functions have been found useful in particular rocket ballistics problems, the one most frequently used being that of the French *Commission de Gâvre*, not so much because of its merit but because most of the available ballistic tables (in particular those usable for high-angle fire from the ground) are based on it. The Gâvre function is based on drag measurements of an obsolete type of shell having a relatively blunt nose and no boattailing, and the fact that many contemporary rockets have these same characteristics provides some justification for its use.

It would take us too far afield to discuss the details of the methods of range calculations. They are given in *Exterior Ballistics*.[1b] In addition, a good bibliography on the subject is contained in *Rocket Fundamentals*.[8a]

24.4.3 Launcher "Tip-off" Effects

In correcting the quadrant elevation of the rocket to that of the equivalent shell [equation (6)], it was assumed that the "tip-off" is negligible. During the short time when the center of gravity of the

[redacted]

rocket is off the launcher but the tail (or rearmost point in contact with the launcher) is still in contact, a torque exists tending to give the rocket an angular momentum about a horizontal axis. In the case of spin-stabilized rockets, the combination of this torque with the gyroscopic effect results in a deflection of the round to the left (assuming right-hand spin), but for a fin-stabilized rocket, tip-off simply reduces the effective launching angle and hence reduces the range if the quadrant elevation is 45 degrees or less. Theoretical analysis [e] shows that the *amount* of rotation *during* tip-off is negligible, but that the angular *velocity* imparted to the rocket persists and continues to decrease the effective elevation angle so that, for rockets which are launched at very low velocities, the total reduction in effective launching angle can amount to several degrees.

Tip-off can be reduced in two ways:

1. By reducing the ratio of burning distance to launcher length, either by using a longer launcher or a grain giving shorter burning time, so that the rocket is launched at higher velocity; or

2. By arranging that the rocket is not constrained after the center of gravity leaves the launcher. This has been accomplished in certain cases by using a special launcher such as the "zero-length" launcher or, for the antisubmarine rocket, by making the diameter of the tail smaller than that of the head so that the tail does not touch the launcher at all.

24.5 WIND EFFECT

The effect of a uniform wind on the flight of a finner follows simply from the fact that the aerodynamic moment is a righting moment. From whatever direction the wind is blowing, its force, being greater on the tail than on the nose, will push the tail downwind so that the rocket will head into the wind. This effect is most striking in the case of rockets fired from aircraft, i.e., in high relative winds. To take an extreme example, suppose that a 5.0-in. *high-velocity aircraft rocket* [HVAR] is fired from an airplane traveling 450 mph pointing 10 degrees away from the resultant wind. Then, if the temperature is low so that the burning time is 1.2 seconds or more, its apparent launching direction at the end of burning will deviate from the wind direction not by 10 degrees but by less than 0.2 degree. In ground firing, the effect is qualitatively similar but much smaller. Consider a wind blowing at right angles to the launcher; then it is obvious that its effect is divided into two parts:

1. During the burning period the action of the wind on the fins will turn the nose into the wind, and the jet will push the rocket in the direction that it points. As long as burning continues, the tangent to the trajectory, although oscillating slightly as shown in Figure 4, deviates on the average farther and farther from its original direction. If a long-burning rocket is launched at very low velocity, it may be pointing almost directly upwind, when it ceases burning, but, in the usual ground-firing case, the turning into the wind amounts to only a few degrees.

2. After burning, the rocket will drift downwind. This drift comes about not, as might be thought, because of the cross-wind force ("lift") but because of the action of the downwind component of the drag. The reason is that the period of oscillation of the rocket is small compared to the total time of flight, and, since the yaw oscillations are about the position in which the yaw and lift are zero, the effect of the lift approximately averages to zero. If the rocket is headed almost directly upwind, then obviously the only effect of the wind is to slow it down. In the usual case where the trajectory makes a large angle with the wind, the tangent to the trajectory turns gradually back toward its original direction and may go beyond it if the flight continues long enough. Whether the resultant deflection is upwind or downwind depends upon the ratio of total flight time to burning time, that is, upon the quadrant elevation and the temperature.

The theoretical expressions for the deflection of the trajectory by a cross wind during burning are derived in reference 11 and in *Exterior Ballistics*,[1] and the results are shown graphically in Figure 4 in terms of dimensionless parameters which can be applied to any fin-stabilized rocket. For any given rocket, the ordinates are proportional to the deflection of the trajectory from the original launcher line, the abscissas are proportional to velocity, and the parameters characterizing the various curves are proportional to the square root of the launcher

[e] The theory of tip-off is derived in two local CIT reports[9,10] for rockets like the 4.5-in. barrage rocket in which a single point on the tail touches the launcher after the head leaves it, and in *Rocket Fundamentals*,[8] for rockets of uniform diameter for which the point of contact with the end of the launcher moves along the rocket. Both cases are treated in *Exterior Ballistics*.[1]

[redacted]

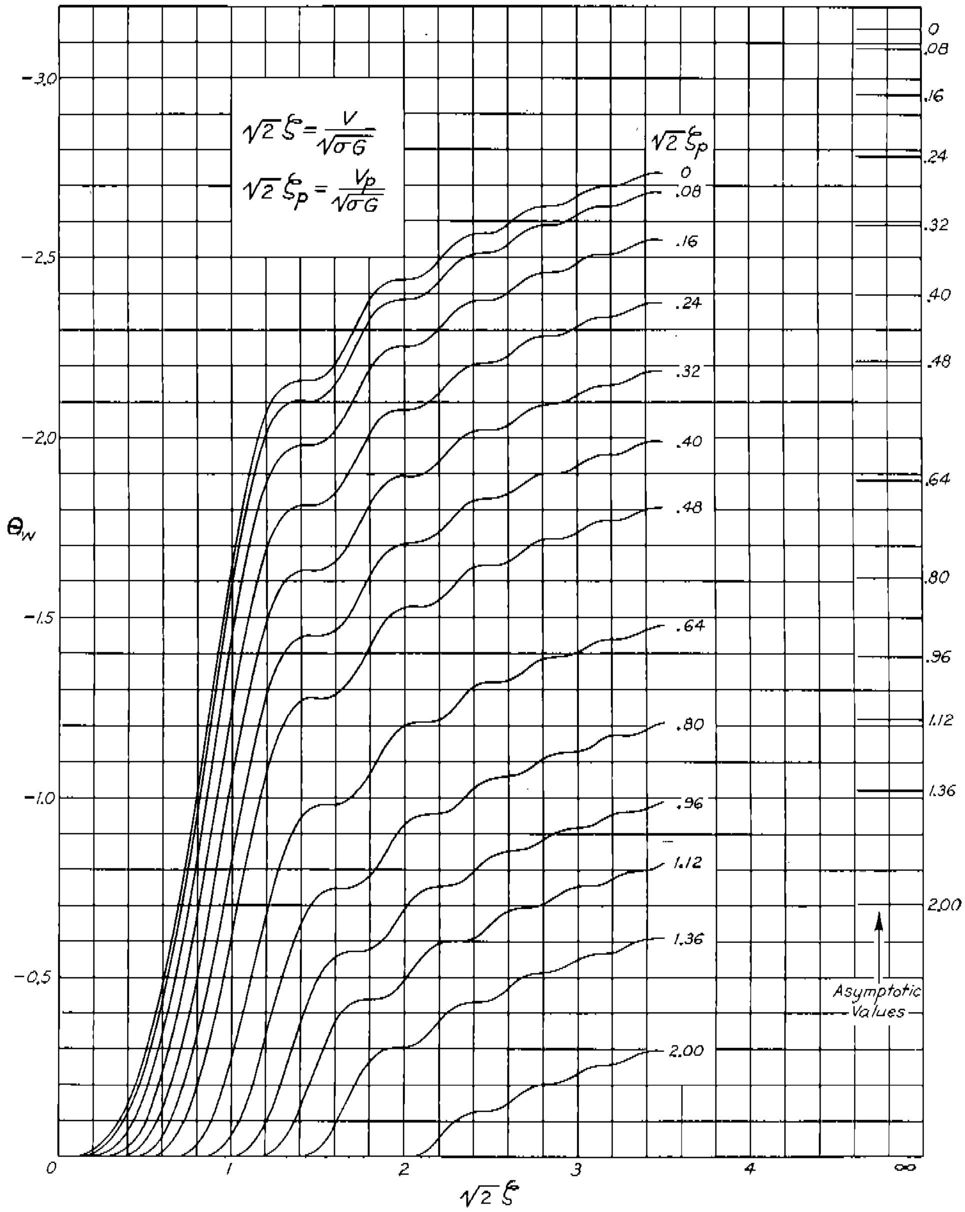

FIGURE 4. Deflection of trajectory by cross wind.

length. The symbols and their relations are as follows:

$\zeta \equiv$ "velocity parameter."

$$\zeta = \frac{V}{V_\sigma}; \quad \zeta_p = \frac{V_p}{V_\sigma} = \sqrt{\frac{p}{\sigma}}.$$

$\Theta_W \equiv$ "characteristic function for trajectory deviation by a cross wind." The actual angle θ in radians of deviation of the tangent to the trajectory from its original direction for any particular rocket is obtained by substituting the proper values into the relation:

$$\theta = \frac{W_N}{V_\sigma}\Theta_W.$$

$\sigma \equiv$ distance traveled by the rocket during one cycle of yaw oscillation (ft).

$G \equiv$ acceleration of the rocket (ft/sec²).

$p \equiv$ length of the launcher (ft).

$V \equiv$ instantaneous velocity of the rocket (fps).

$V_p \equiv$ velocity with which the rocket leaves the launcher (fps).

$V_\sigma \equiv$ velocity of the rocket at the end of the first yaw oscillation cycle (fps).

$$V_\sigma = \sqrt{2G\sigma}.$$

$W_N \equiv$ component of wind velocity perpendicular to the launcher (fps).

[redacted]

By inserting into the graph the proper value of ζ, we can calculate the trajectory deviation at any time during burning or at the end of burning.

To see the importance of the wind effect in ground firing, let us consider the effect of wind on the 4.5-in. barrage rocket at 70 F, and at the end of the burning period. For this

$$G = 960 \text{ ft/sec}^2;$$
$$\sigma = 265 \text{ ft};$$
$$V_\sigma = 715 \text{ fps};$$
$$p = 5 \text{ ft};$$
$$\sqrt{2\zeta p} \approx 0.2;$$
$$V_b \equiv \text{ velocity at end of burning} = 355 \text{ fps};$$
$$\zeta_b \equiv \zeta \text{ at end of burning} \approx 0.5.$$

Reading the value $\Theta_W = 0.5$ from the graph (the negative sign simply means that the deflection is upwind), we calculate for the trajectory deflection per unit cross wind the value of 0.7 mils per fps. Since the lateral dispersion (mean deviation) of the rocket is about 45 mils, it will apparently take a rather large side wind to change the center of impact by an amount comparable to the dispersion, especially since part of this deflection is canceled out by the drift after burning. The actual effect of wind on the impact point of the barrage rocket is given in Table 2.[f]

TABLE 2. Wind deflection of impact point for 4.5-in. barrage rocket.

Propellant temperature (°F)	Angle of elevation (degrees)	Lateral deflection *into* the wind for wind of 1 mph (yd)	Increase in range for tail wind of 1 mph (yd)
40	20	0.7	1.6
40	45	1.4	1.6
70	20	0.2	1.2
70	45	0.5	1.7
100	20	−0.1	1.0
100	45	−0.1	1.6

The effect on the trajectory of the component of wind in the direction of the launcher is virtually negligible, so that the general case of wind in any direction is obtainable from the curves of Figure 4. Thus the lateral deflection on the horizontal plane is obtained by inserting the component of wind perpendicular to the line of fire and dividing the result by the cosine of the quadrant angle. The change in effective launching angle by up-range and down-range winds is given by using as W_N the along-range component multiplied by the cosine of the quadrant angle.

The calculation of the drift after burning can be done by simple methods familiar in artillery theory. They are discussed in references 11 and 13.

[f] A more complete table is included in reference 12.

24.6 TRAJECTORIES OF ROCKETS FIRED FORWARD FROM AIRPLANES[g]

By far the most extensive application of external ballistic theory to rockets has been in connection with forward firing from aircraft, for it is only in this use that fin-stabilized rockets are sufficiently accurate to warrant accurate calculations of trajectories. As in the case of ground trajectories, the solution requires setting up the equations of motion of the rocket in the air and integrating them, but the solution is simpler here because of the relatively short flight times that have been used in practice. The methods of calculating the trajectories and the sighting tables required for various aircraft and various firing conditions are worked out in detail in reference 15 and in *Exterior Ballistics*.[1]

The characteristics of the rocket trajectory can best be understood through a comparison with those of the more familiar machine gun bullet.[h] If firing conditions are identical, the two trajectories differ mainly in the following three respects:

1. *Rockets are slower.* The velocity of the fastest rocket used at present in forward firing is approximately 1,350 fps, which is about half that of a .50-caliber machine gun bullet. Furthermore, it takes the rocket a relatively long time—of the order of 1 second, more or less depending upon the temperature—to reach its maximum velocity, whereas the bullet has its maximum velocity as it leaves the muzzle. The consequent longer time of flight of the rocket to a given range means that allowances for target speed and wind are much greater than in the case of machine gun fire.

2. *Rockets tend to follow the direction of flight of the aircraft, whereas bullets travel in the direction of the gun.* The bullet travels close to the direction of aim because its muzzle velocity is so much greater

[g] A basic reference on this subject is *Firing of Rockets from Aircraft*,[14] one of the CIT final reports under OEMsr-418.

[h] See reference 16 for an excellent simple discussion of the general discussion of the general features of aircraft rocket trajectories.

[redacted]

than the speed of the airplane that the effect of the latter upon the motion of the projectile is relatively slight. We have already seen that the fins of a rocket, on the other hand, tend to align it with the airflow resulting from the combination of the velocities of rocket and aircraft. Since the launching speed is low, the rocket is quickly aligned almost in the direction of the line of flight of the aircraft. This deflection toward the flight path is greater the less the launching speed of the rocket, and is almost 100 per cent with post launchers.

3. *The rocket trajectory is characterized by considerable curvature compared with the flat trajectory of a bullet.* Not only does the longer time of flight lead to a greater gravity drop, but, in addition, as the rocket falls, the fins tend to align it along the trajectory so that there is also a component of the jet force downward contributing to the normal gravity drop. The consequent large curvature means that the sighting allowance required in aiming and its variation with dive angle are much greater for rockets than for guns.

24.6.1 Trajectories of Post- and Rail-Launched Rockets

The basic object of trajectory calculations for aircraft rockets is obviously to establish the relation of the position of the rocket at a given range to the position of the aircraft sight. The analytical expression for the trajectory may contain three terms: (1) the gravity term, (2) the yaw term, and (3) the angular velocity term which is very small for firing from fixed launchers (either post or rail type). The values of the terms as functions of propellant temperature, airspeed, dive angle, and slant range have been calculated for the various aircraft rockets and published in a number of reports.[i]

The trajectory drop (gravity term) is shown as the distance CE in Figure 5. It depends on:

1. The rocket type, being smaller for higher velocity rockets;

2. The propellant temperature, being smaller for higher temperatures because of the decreased burning time;

3. The dive angle, varying approximately as the cosine of this angle because of the different effective direction of gravity relative to the flight line;

4. The launching speed, decreasing with higher speed; and

5. The slant range to the target.

The yaw term consists of the product of the initial yaw and a "launching factor." When a rocket is fired into the airstream with an initial yaw to the stream, its trajectory is turned toward the direction of the relative wind by the action of the fins and the jet. The ratio of the angle through which the trajectory turns to the initial angle of yaw is called the launching factor f and may have values from 1.00 down to less than 0.70, depending on the rocket type, the propellant temperature, the length

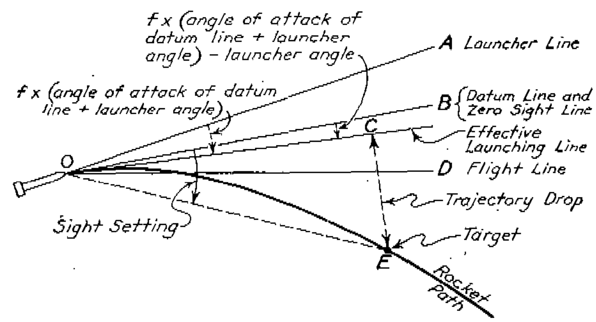

FIGURE 5. Factors in sighting for forward firing.

of constrained motion on the launcher, and the indicated airspeed of the airplane. The evaluation of the initial yaw is complicated because the angle of attack (angle between flight line and boresight datum line [BSDL]) depends upon so many factors—airspeed, dive angle, gasoline load, bomb load, etc. In addition, if there is a side wind, the plane axis will make an angle with the flight line in a horizontal plane, introducing horizontal as well as vertical yaws. Were it not that f is usually very close to 1.0, the problem of firing rockets from aircraft would be even more complicated.

The angular velocity term is similar to the yaw term. If the rocket enters the airstream with an initial angular velocity, its trajectory is deflected in the direction of the angular velocity, and the ratio of the angle of deflection to the initial angular velocity is defined as the angular velocity factor. Its value varies from about 0.25 virtually zero.

If angular velocity is negligible:

Sight setting = trajectory drop $+ f \times$ (angle of attack of datum line + launcher angle) − launcher angle

= trajectory drop $+ f \times$ angle of attack of datum line $- (1 - f) \times$ launcher angle.

[i] See UBC reports listed in the CIT OEMsr-418 bibliography in the general bibliography in the appendix.

[redacted]

In case the launcher angle (i.e., the angle between the launcher and the zero sight line) is zero, only the first two terms in the sight-setting equation occur. For most aircraft, the fact that the sight is separated from the launchers by several feet adds the term h/R to the sight setting, where R is the slant range expressed in thousands of yards and h is the distance between the zero sight line and the launcher line expressed in yards. Sight settings are customarily expressed in mils.[j]

24.6.2 Angle of Attack

The most uncertain quantity in ordinary forward-firing problems is probably the angle of attack, the angle which the boresight datum line makes with the line of flight of the aircraft. It is necessary to differentiate between the *true* angle of attack and the *effective* angle of attack. The former is the angle between the BSDL and the *undisturbed airflow* at a great distance from the airplane. For simplicity in calculation of trajectories, it is customary to assume a uniform airstream around the airplane, although the direction of airflow adjacent to the airplane actually bears very little relation to the flight line, the effects of the propeller, fuselage, and especially the wings resulting in a flow which is uniform neither in magnitude nor direction. To circumvent this difficulty one defines the effective angle of attack to be that angle which gives the right answer in the sight-setting equation and then determines it experimentally for each aircraft under various flight conditions. The prediction of effective angles of attack is an exceedingly difficult problem. For example, it was found that firing the 5.0-in. *aircraft rocket* [AR] with and without the 11.75-in. AR mounted in the airstream produced effective angles of attack differing by 10 mils, even after corrections for the differences in weight had been made. The problem is discussed in considerable detail in *Firing of Rockets from Aircraft*,[14] and an attempt at a theoretical understanding of it is made in reference 17 and in *Exterior Ballistics*.[1]

[j] At least two definitions of a mil are current. The standard Army mil is 1/6400 of a complete circle or 0.056250 degree, but in theoretical discussions it is more convenient to use the milliradian, 0.057296 degree. The latter is 1.86 per cent larger than the former, but, for practical purposes when small angles are involved, either may be taken as 1-yd deflection in 1,000-yd range.

24.6.3 Displacement and Drop Launchers

The calculation of trajectories and sight settings for other launching methods involves no essentially new concepts and hence will not be discussed here. Because the initial conditions are more complicated, the sight-setting equations contain several more terms. The reader should consult *Exterior Ballistics*,[1] *Firing of Rockets from Aircraft*,[14] or reference 18.

24.7 RETRO FIRING

Firing fin-stabilized rockets backwards from aircraft is no longer of much interest and will not be discussed here. Some ballistic calculations on the problem are given in reference 19. Firing fin-stabilized rockets accurately in any other direction is obviously impossible because of the large f factor.

24.8 DISPERSION OF FIN-STABILIZED ROCKETS

Dispersion is a measure of the scatter of the impact points of a group of identical rockets fired under supposedly identical conditions. Ordinarily this scatter is measured about the mean impact point of the group, but in some cases it may be measured about the point which one assumes would be the mean impact point of a very large group of rounds; e.g., the lateral dispersion may be measured about the range line. Several different quantitative measures of dispersion are in current use, some of which are illustrated in Figure 6 (Figure 9 of reference 20). For our purpose we shall adopt the mean deviation as the measure of dispersion and shall use the two terms interchangeably. The mean deviation is computed by adding the deviations of the various rounds from the mean without regard to algebraic sign and dividing by the total number of rounds. Lateral dispersion is usually expressed in mils or in yards, and range dispersion in per cent of mean range or in yards.

Many factors contribute to dispersion. Thus range dispersion may be introduced by variations in rocket weight, propellant weight, or effective gas velocity among different rounds of the group. Both range and lateral dispersion are affected by variations in burning time, propellant temperature, or wind velocity and by irregularities such as rough or crooked launchers, misaligned fins, and faulty lug bands. It was shown very early in the OSRD rocket developments, however, that finners fly quite straight after the cessation of burning and that the nozzle axis—and "gas malalignment"—the malalignment remaining when the mechanical malalignment is zero. No way is known to measure the gas malalignment directly, and it is usually inferred from an experimental test of dispersion by subtract-

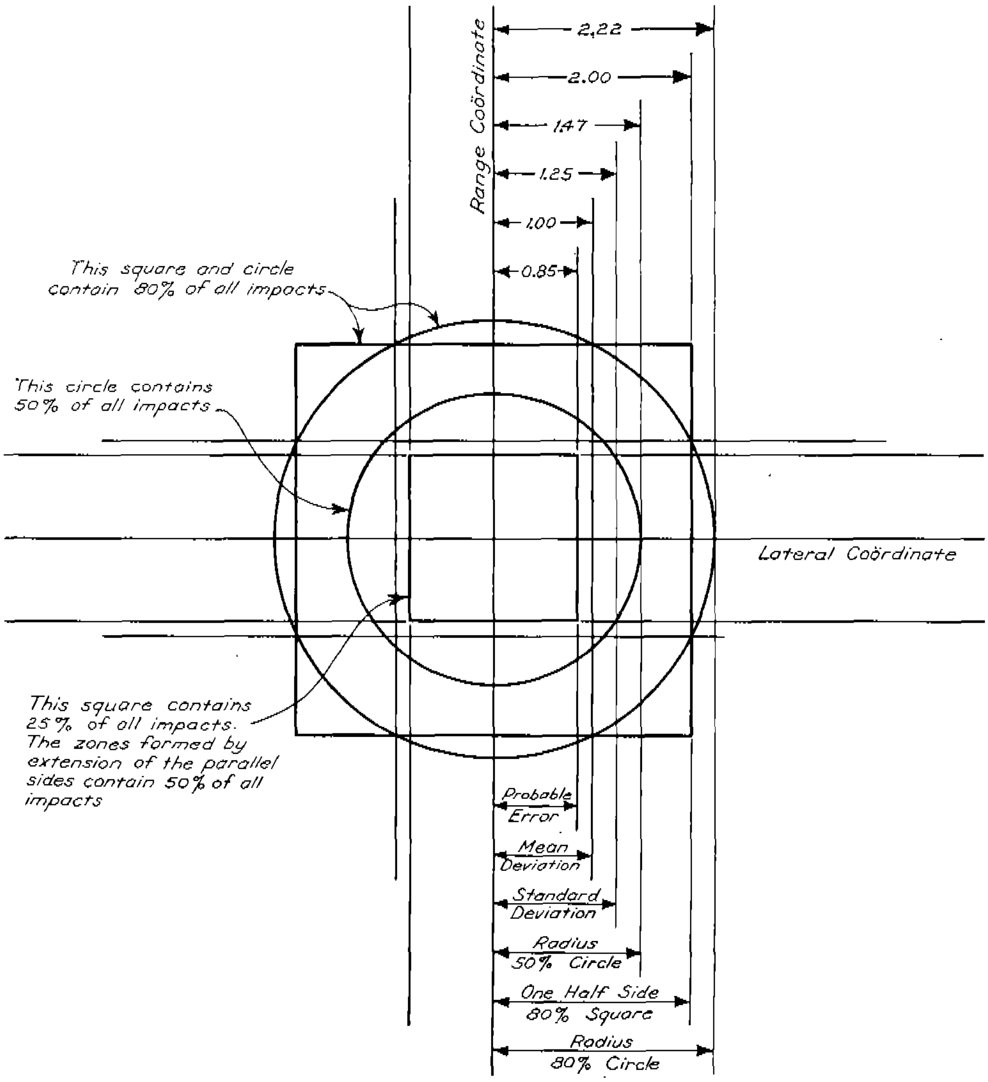

FIGURE 6. Comparison of various measures of dispersion.

predominant cause of dispersion during burning is the malalignment.

The malalignment of a rocket is usually defined as the distance between the center of mass of the rocket and the line of action of the thrust. Since the line of thrust coincides approximately with the geometrical axis of the exit cone of the nozzle, a distinction is made between "mechanical malalignment"—the distance of the center of mass from the ing the effect of known mechanical malalignment. This procedure necessarily lumps together as gas malalignment all the other errors which can contribute to dispersion, so that the result is always too large by an unknown amount. It is known, however, that random variations of the thrust direction from the nozzle axis occur during burning, and the concept of gas malalignment is useful even though not precise.

[redacted]

24.8.1 Dispersion of Finners in Ground Firing

Whatever the cause or type of malalignment, its effect is a torque which causes the rocket to yaw and hence to deviate in the direction of that yaw. The resulting dispersion has been discussed qualitatively in Chapter 21 and was first expressed quantitatively in *The Effect of Fin Size, Burning Time, and Projector Length on the Accuracy of Rockets*,[21] a report which has become a classic in rocket literature.[k] In this analysis, damping, drag, gravitational force, and cross-wind force are assumed negligible, and the equations of motion of the projectile are solved assuming the malalignment torque to be constant and the restoring torque of the fins to be proportional to the yaw and to the square of the velocity. The solutions of the equations turn out to be Fresnel integrals, and they are plotted in Figure 7 from which the qualitative conclusions listed in Chapter 21 and many others, can be deduced. If one considers a particular projectile, the ordinates in Figure 7 are proportional to deflection of the trajectory in the plane of the yaw per unit malalignment, the abscissas are proportional to time, and the parameter is essentially projector length. Thus each curve shows the variation in trajectory deviation (the angle θ in Figure 1) with time during burning, and, since the rocket, after burning continues its flight in the direction it was pointing when the thrust ceased, putting the value t_b into the graph gives the trajectory direction at all times after burning. To convert this to lateral deviation of the impact point, which is most frequently of interest, one must multiply by the sine of the angle between the plane of yaw and the vertical plane and divide by the cosine of the angle of elevation of the launcher, the small corrrection for the burning distance usually being neglected. It will be noted that two abscissa scales are included, the upper one applying at all times and the lower one being appropriate only at the end of burning.

Despite the seemingly rather restrictive assumptions on which the theory is based, it has been found to be in excellent agreement with experiment. The fact that the malalignment torque is not constant either in magnitude or direction during burning does not invalidate the conclusions with regard to variation of dispersion with launcher length, burning time, or fin size (i.e., the yaw oscillation distance σ, defined in Section 24.5 and in Chapter 21). In particular it is important that the theory holds fairly well even for supersonic velocities where the restoring moment is not proportional to the square of the velocity, because normally a rocket reaches the flat portion of the curve before the square law breaks down so that practically all of its deflection is acquired in the low-velocity region where the theory is valid.

The chief limitations of the theory are the difficulty of determining the effective launcher length p (p stands for "projector") and the actual malalignment L_0 during flight. An accurate definition of p would be the distance through which the rocket is constrained to move with zero deflection, but whether this constraint ceases when the head or front lug leaves the launcher or continues as long as the center of mass, the tail, or the rear lug is in contact is seldom obvious a priori. Analysis of a large number of firings of the 4.5-in. barrage rocket, for example, showed that the data could best be brought into agreement with the theory by assuming that the constraint ceases when the tail leaves the launcher.[23] Probably this is approximately the case for most relatively lightweight rockets on rail or tubular launchers.

On the other hand, the 5.0-in. HVAR was found to be about equally accurate from zero-length post launchers or the 7½-ft Mk 4 launcher even in ground firing. The curves of Figure 7 provide the explanation[24] of the behavior of the HVAR. This rocket has an initial acceleration of $55g$, from which it is easily calculated that its rear lug will clear a 7.5-ft launcher in approximately 0.09 second. Since its effective burning time at 70 F is 0.9 second and its yaw oscillation distance σ is 320 ft, the values of $t\sqrt{G/\sigma}$ which we need for the graph are 2.1 at the end of burning and 0.21 at 0.09 second, and the ordinates of the curve $p/\sigma = 0$ corresponding to these times are respectively 0.010 and 0.0006. Thus, if fired from a zero-length launcher, this rocket will acquire 0.06 of its total deflection during burning in the first 7.5 ft. If its average deflection at the end of burning is 20 mils, then at 7.5 ft it will be 1.2 mils. Since the separation between the two suspension lugs is approximately 36 in., they will undergo a relative lateral displacement of 0.043 in. in the first 7.5 ft. *But the clearance between the lugs and the slot of a Mk 4 launcher is approximately 0.060*

[k] There are several earlier CIT reports on the same subject. A later one[22] includes simplified formulas useful in restricted regions.

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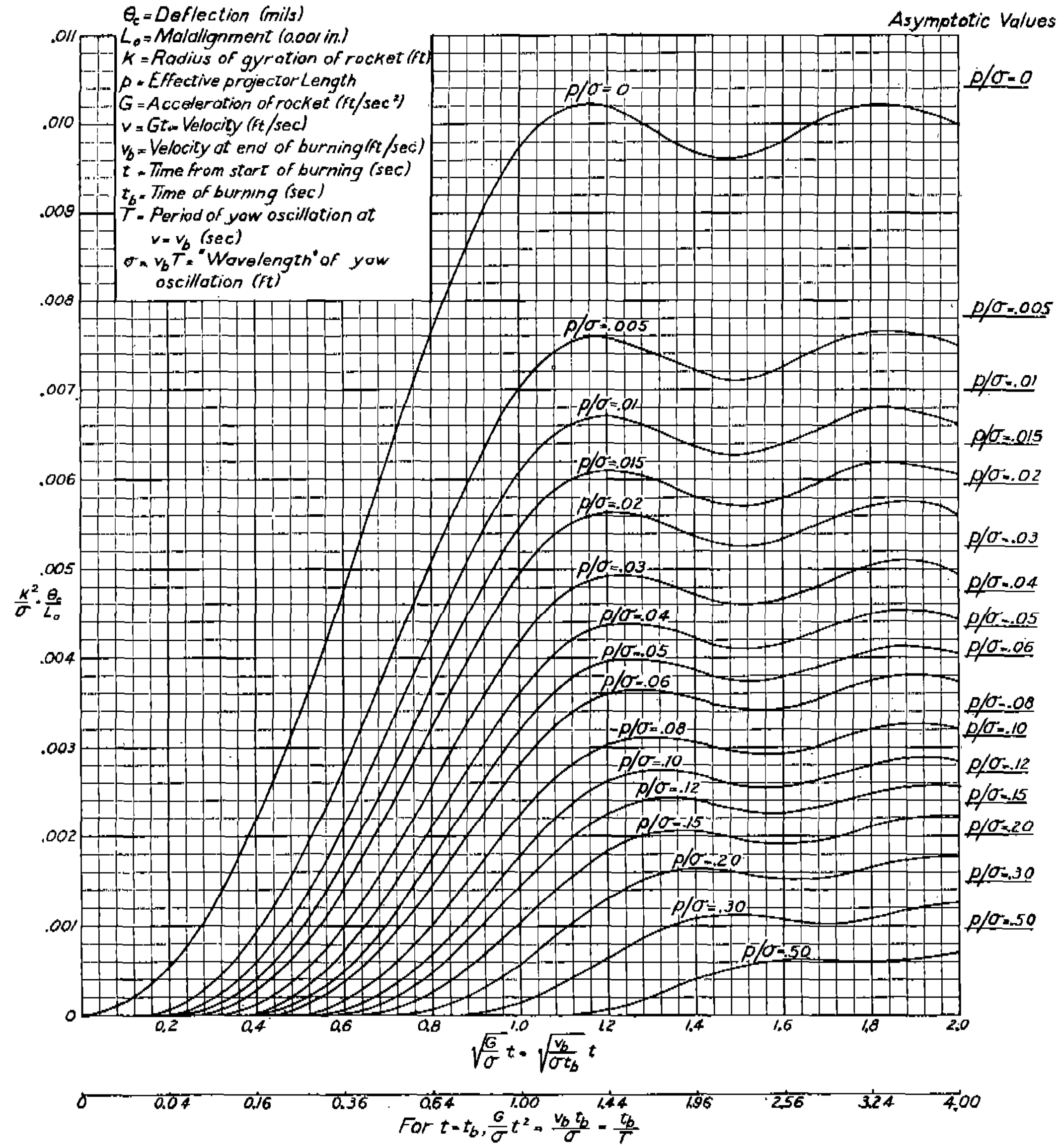

FIGURE 7. Deflection of trajectory by malalignment (zero launcher velocity).

in. Hence it is clear that it subjects the average round to little or no constraint and its effective length must be almost zero. Even if the clearances were made quite small, it is doubtful that the dispersion of a rocket as heavy as the HVAR would be improved, because it is not feasible to build an aircraft launcher of sufficient weight and rigidity to constrain it effectively.

More detailed applications of the theory to various CIT rockets are given in references 25 and 26, and one more will be included here. The *chemical warfare rocket* [CWR] (see Section 18.4), before the addition of radial fins, had the following characteristics:

Velocity at the end of burning (70 F): $V_b = 710$ fps;

Yaw oscillation distance: $\sigma = 280$ ft;

Radius of gyration (see Table 2 in Section 21.3): $K = 1.22$ ft.

Hence

$$\frac{K^2}{\sigma} = 0.0053;$$

$$\frac{V_b}{\sigma} = 2.54.$$

[redacted]

Assuming an effective launcher length $p=8.4$ ft and a burning time $t_b=0.51$ second, we have

$$\frac{p}{\sigma} = 0.03;$$

$$\frac{V_b t_b}{\sigma} = 1.3.$$

From Figure 7[1] we read

$$\frac{K^2}{\sigma} \cdot \frac{\theta}{L_0} = 0.0053; \frac{\theta}{L_0} \approx 0.0048.$$

Hence

$$\frac{\theta}{L_0} \approx 0.9 \text{ mils per 0.001-in. malalignment.}$$

Increasing the burning time above this value would make no significant change in dispersion, since all values of $(K^2/\sigma)(\theta/L_0)$ for longer burning times fall between 0.0046 and 0.0053. Reducing the burning time by half, on the other hand, giving

$$\frac{V_b t_b}{\sigma} = 0.64$$

results in

$$\frac{K^2}{\sigma} \cdot \frac{\theta}{L_0} \approx 0.0026,$$

so that

$$\frac{\theta}{L_0} = 0.49 \text{ mils per 0.001-in. malalignment.}$$

It is easily seen also from the graph that the same improvement in dispersion without reduction in burning time could be obtained with $p/\sigma=0.09$, that is, triple the original effective launcher length, if such a launcher were practicable.

The theory is primarily useful for making comparisons of this type, but it can be used to predict the actual magnitude of dispersion if something is known about the average malalignment to be expected in practice. From measurements on various rockets it is known that the minimum attainable gas malalignment is approximately 1 mil and values of 2 or 3 mils are common. Since the CWR has its center of mass 25 in. ahead of the nozzle throat, 1 mil corresponds to a malalignment of 0.025 in. Using this value and $\theta/L_0=0.9$, we find for the dispersion expected at 45 degrees elevation angle

$$\frac{0.9 \times 25 \times 2}{\pi \cos 45°} = 20.3 \text{ mils.}$$

The factor $2/\pi$ occurs because the directions of the malalignments are randomly distributed. One would expect 20 mils to be an extremely optimistic estimate of dispersion, and in fact the actual dispersion is nearer 45 mils, indicating that 2 mils would have been a better guess at the gas malalignment. If the rockets are not well made, the mechanical malalignment may further increase the dispersion, but with careful manufacturing and inspection it is usually possible to keep the mechanical malalignment small enough so that its effect is completely obscured by the gas malalignment.

The theory indicates the following possible ways to increase the accuracy of a fin-stabilized rocket:

1. Increase the launcher length; when it can be done, this will always reduce the dispersion if the rocket is actually constrained by the launcher, but it is seldom practicable.
2. Decrease the burning time; this will be effective only if it can be decreased to the point where it is considerably less than the period of yaw oscillation σ, which is seldom possible for high-performance rockets.
3. Reduce σ; i.e., increase the stability by using larger fins or (usually less feasible) by moving the center of mass forward. The difficulty here is that rather large increases in fin size are required to affect σ appreciably (see Chapter 19 for the effect of fin size on the HVAR), and these are usually precluded by space considerations.
4. Increase the radius of gyration K; in other words, design a new rocket that is longer and slimmer.
5. Reduce the gas malalignment.

Much effort has been expended in attempts to reduce gas malalignment by some variation in the interior of rocket motors. A number of the expedients tried are discussed in reference 27. None showed any promise of success. Apparently gas malalignment is rather fundamentally tied in with high-speed gas flow and cannot be significantly reduced. Its effect can be partially circumvented, however, by two expedients: using multiple nozzles and rotating the rounds. Apparently the directions of gas malalignment in various nozzles of one plate are at least partially independent and tend to cancel

[1] The subscript c on the θ's in Figure 7 signifies merely that it was calculated on the assumption of constant acceleration and constant malalignment.

[redacted]

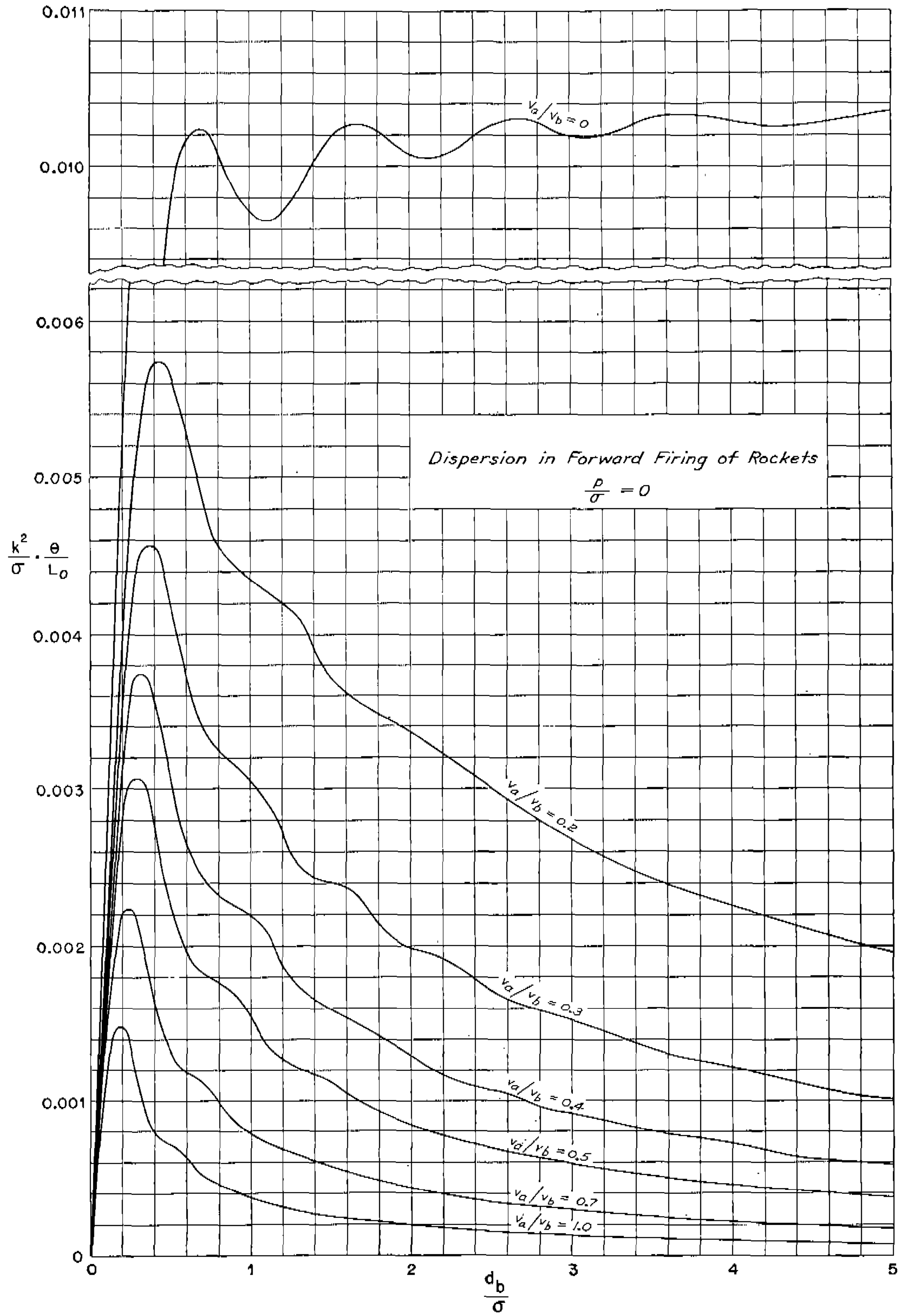

FIGURE 8. Deflection of trajectory by malalignment in aircraft forward firing from "zero-length" launchers.

[redacted]

one another out. Thus the gas malalignment of the eight-nozzle HVAR is less than 0.87 mils.[28] Except for the very atypical target rocket, no serious attempts to reduce dispersion by rotating fin-stabilized rockets were made by CIT. The British and the Section H[29] workers have tried it with some success, however, and the theoretical possibilities along this line are discussed briefly in *Exterior Ballistics*.[1c]

24.8.2 Dispersion in Firing Finners Forward from Airplanes

By a simple change of variable, the curves in Figure 7 can be adapted to the case where the rockets have a relative velocity with respect to the air at the beginning of burning, thus giving the dispersion caused by malalignment in forward firing from aircraft. This theory is derived in reference 30 and reduced to graphical form in reference 31. Since one more parameter, the airspeed of the plane, now appears in the theory, it is not possible to show the whole story on a single graph, and only the curves for the most important case, zero launcher length, are reproduced here as Figure 8. An examination of the complete set of curves leads to the following conclusions:

1. For given values of p/σ and V_a/V_b (aircraft velocity divided by rocket burnt velocity relative to the aircraft), the dispersion increases rapidly with burning distance d_b, reaching a maximum at a value of d_b/σ between 0.2 and 0.5 and then decreases (except for extremely small V_a) for longer burning distances.

2. Dispersion decreases with increased launcher length, but this effect becomes less marked at higher airplane speeds. Thus for the HVAR ($V_b = 1{,}350$, $d_b \approx 600$, $\sigma \approx 300$), theory predicts the following decreases in dispersion in going from a zero-length to a 6-ft launcher:

Ground firing	44 per cent;
Airspeed 270 fps	41 per cent;
Airspeed 540 fps	36 per cent.

In practice, the improvement is likely to be considerably less than this, as pointed out in the previous section.

3. The relative gain in accuracy when going from a stationary to a moving launcher is greatest for the zero-length launcher, where it can amount to a factor of 10 or more. The burning distances of CIT aircraft rockets are all so long as to place them well beyond the maxima in Figure 8. In this case, asymptotic formulas are applicable, and the single convenient curve of Figure 9 (taken from reference 32) covers all cases.

The dispersion of the ammunition itself is by no means the predominant effect in forward firing, however. Many other factors contribute to the inaccuracy, such as sighting errors, faulty estimation of range to the target, airspeed and dive angle incorrect for the sight settings used, uncertainty in the temperature of the ammunition, random winds, and firing while the dive angle is changing. The quantitative effects of these various errors are analyzed in reference 33, from which is taken Table 3, showing a typical case. Many good illustrations of these effects are given in reference 16.

TABLE 3. Effect of various factors on dispersion in forward firing of 3.5-in. AR from TBF-1.

Launcher 3° above datum line — Dive angle 20°
Mean temperature 70°F — Range 750 yd
Launcher length 7.5 ft

Vertical dispersion (mils)				
Aircraft speed (knots)	200	225	250	275
Ammunition dispersion	5	4	3	2
Pure aiming error	3	3	3	3
Random wind (10 fps)	3	3	3	3
Range error (75 yd)	2	2	2	2
Temperature error (10°F)	2	2	2	2
Aircraft speed error (10%)	6	6	6	6
Total	9	9	8	7
Lateral dispersion (mils)				
Ammunition dispersion	5	4	3	2
Pure aiming error	2	2	2	2
Wind (10 fps)	8	8	8	8
Total	10	9	9	8

24.9 UNDERWATER TRAJECTORIES

The underwater ballistics of fin-stabilized rockets has already been briefly introduced in Chapter 15 in connection with head shapes. We have seen that the projectile after entering the water travels in a bubble and is in contact with the water only near the nose and the tail. In this position it effectively has a yaw with its trajectory; consequently the forces of the water reacting on the nose are not in general symmetrical, and a net cross force exists on the nose. In the case of a pointed projectile this cross force is in the direction opposite to the side of the bubble on which the tail lies, and hence is

[redacted]

usually an upward force because both the effect of gravity and the initial impulse of the water on the nose tend to make the tail ride on the bottom of the bubble. It has been demonstrated that the amount of this cross force varies greatly with the shape of the ogive. Thus there should be practically no side force on a hemispherical ogive, since it presents the same form to the water when rotated through a small angle. There will still be a side force on the tail, but, for a long slender projectile in which the center of gravity is near the ogive, this should be negligible. On the other hand, for sharper ogives the side force is greatly increased, since for a given yaw the ratio of the amount of water forced to one side to the amount forced to the other side of the projectile is greater the longer the ogive.

Two other factors in addition to nose shape determine the magnitude of the cross force. The length is important because a short rocket will have to have a larger yaw in order to ride on the bottom of the bubble than will a longer rocket of the same diameter and head contour. Thus an ordinary shell, whether spinning or not, is so short that it cannot be stabilized under water at all, but turns sideways and comes to rest almost immediately. The effect of a motor of diameter less than that of the head is to increase the yaw, because the smaller motor must dip farther into the side of the bubble in order to acquire a given restoring moment. Thus for the 5.0-in. aircraft rocket which has a 3.25-in. motor, no head shape was found which would make the rocket stable under water.

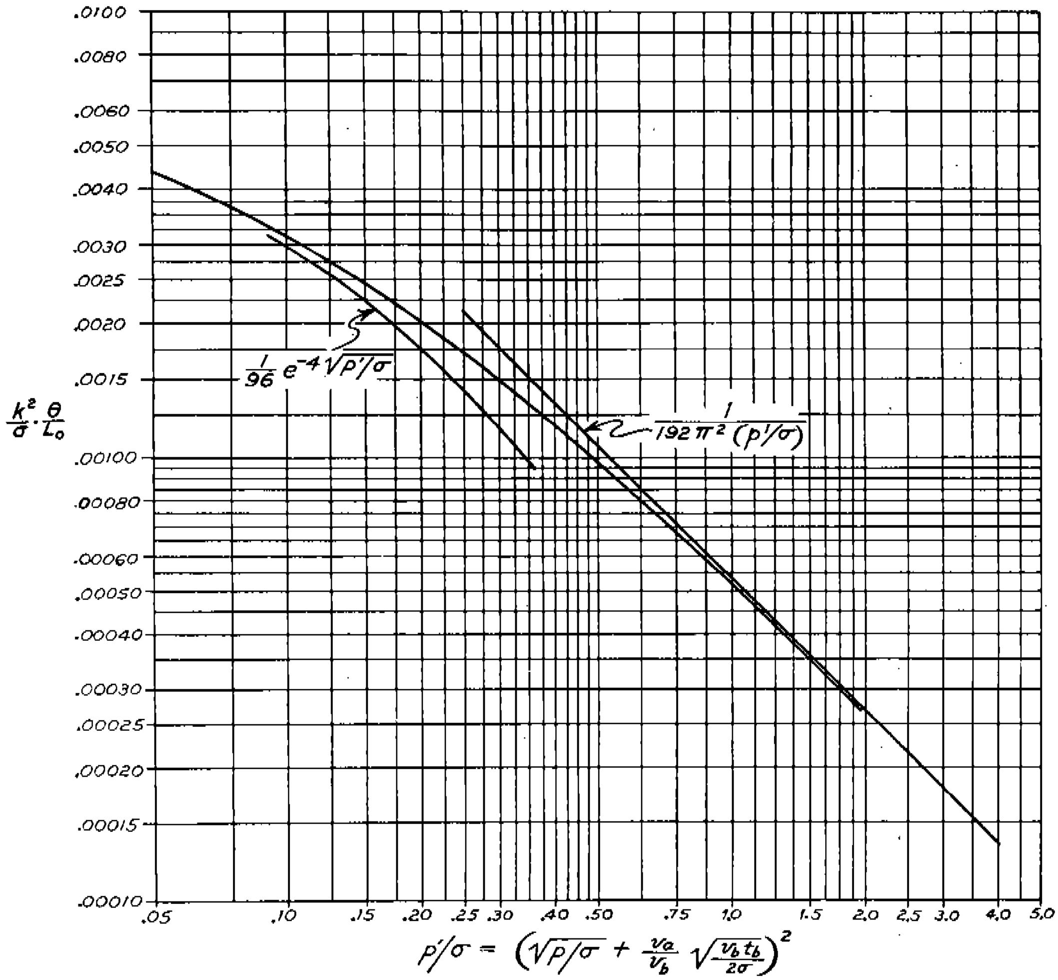

FIGURE 9. Dispersion of long-burning rockets in forward firing. Middle curve is exact for very long-burning rockets; side curves illustrate approximate formulas.

[redacted]

If the side force becomes too great, as it may at high entry velocities and large entrance angles, the rocket breaks in two, usually at the junction between the motor and the head, and the head is brought to rest almost immediately. Otherwise the side force produces a curvature of the trajectory, and it is easily shown that the path approximates an arc of a circle, the radius of which is directly proportional to the rocket's mass and inversely proportional to the cross force.

If this simple picture were always exactly reproduced in practice, every rocket would follow an upward-curving path and have a trajectory as shown in Figure 10 until its velocity was so reduced that gravitational forces became appreciable. If

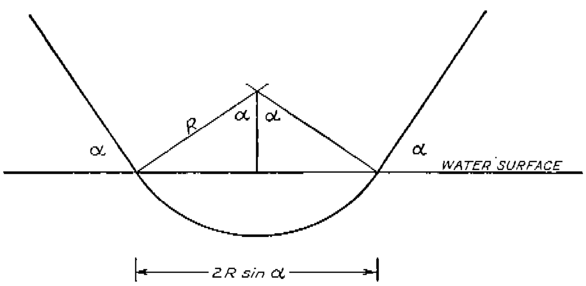

FIGURE 10. Ideal underwater trajectory of a fin-stabilized rocket, assuming negligible change in velocity while below the surface.

fired so as to enter the water at a sufficiently small angle with the surface, it would emerge making the same angle, and the horizontal distance between the entrance and exit points would be proportional to the sine of the entrance angle. Although the limits of error are necessarily rather large, the experimental firings indicate that the average rocket does have such a trajectory. Also in accordance with the theory, it has been found possible to control the radius of curvature within the limits where the rocket can stand the cross force and to reduce the deceleration coefficient substantially by shaping the heads so that the water breaks away from them at a smaller diameter and forms a smaller bubble as discussed in Chapter 15.

Nevertheless, very erratic behavior is exhibited by a small percentage of the rounds, and little is known about the reasons for it. One would surmise that a yaw at the instant of water impact might throw the rocket to one side of the bubble and thus cause the normal curvature of the trajectory to take place in a plane inclined to the vertical. British firings under conditions which allowed recovery of the rounds showed that motor tubes (with thinner wall than American designs) sometimes become distorted by the impact forces and that occasionally one of the four fins remains on the motor; in either of these cases a steering action on the rocket results. A bizarre example of what kinds of things may happen was provided by a Tiny Tim which ricocheted apparently normally and landed on shore, but when recovered was found to have a 1-ft length from the front of the motor tube missing, the head being jammed back into the remaining tube and in fairly good alignment.

24.9.1 Tactical Effectiveness of Underwater Rockets

The ability to vary both the curvature of the trajectory and the rate of loss of velocity under water makes possible a significant increase in the effectiveness of rockets with certain head shapes under certain conditions. A brief quantitative discussion of this point is contained in reference 34 from which all of the theory of underwater trajectories has been taken. Additional theory is discussed in references 35 and 36. Qualitatively, it is evident that the curvature of the trajectory under water causes a deflection from the straight-line air trajectory, which in certain cases may send the rocket into the target, thus increasing the probability of a hit, but in other cases may send it away from the target, decreasing the probability. The rapid deceleration of the rocket under water causes it rapidly to drop below a velocity at which it can cause significant damage, and this factor, as well as the curvature of the trajectory, must be evaluated to determine the rocket's effectiveness under various conditions. For example, consider the case of a submerged submarine, represented in Figure 11 by the circle GHI where the water surface is DEF. ADG and CFI are the extreme trajectories, having an entrance angle α, that just reach the target. The plane MN is perpendicular to the air part of the trajectory. The effective target area then extends from J to L and is significantly wider than the actual target, if the underwater path FI is short enough so that the rocket reached I with a velocity great enough to cause significant damage. If, however, the velocity at I is below that specified to produce the desired damage, a third trajectory must be laid out such that the underwater part of it is equal to the length of underwater travel required to bring the rocket down to the limiting velocity for damage, and the effective target area (proportional to the distance between AD and the air part of this new trajectory) will be correspondingly reduced. The interrelation of these various factors makes the choice of the optimum head shape and entrance angle a rather difficult one, depending very critically on the type of target.

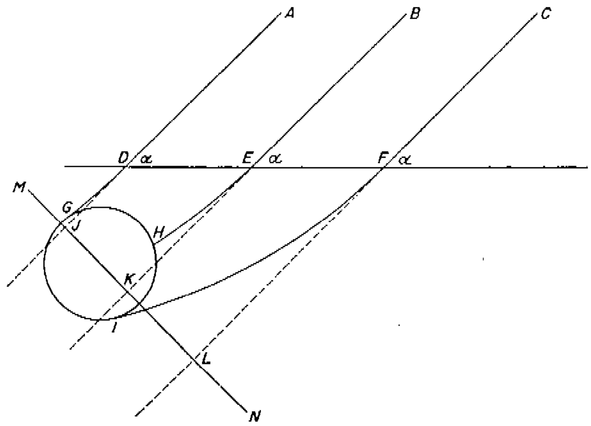

FIGURE 11. Effective target area for submerged cylindrical target.

24.10 UNDERGROUND TRAJECTORIES

Firings of 5.0-in. HVAR's and 11.75-in. AR's into earth have provided additional verification of the theory of underwater trajectories, since one would expect underground and underwater performance to be qualitatively similar. That a rocket travels under ground in a "bubble" is apparent from the erosion marks exhibited by recovered rounds (see Figures 13 and 14). Thus heads which have long straight underwater trajectories (small cross force) actually do give superior performance under ground.

Because of the variable consistency of earth and the meagerness of the data, it is difficult to make any general statements about underground trajectories other than that the much larger forces require heads giving less nose lift than is usable under water. Heads with little or no lift may still be unsatisfactory, however, if their drag is large so that the axial force on the motor is increased. For any head shape, it is essential that the motor tube have a relatively thick wall and that its junction with the head be strong.

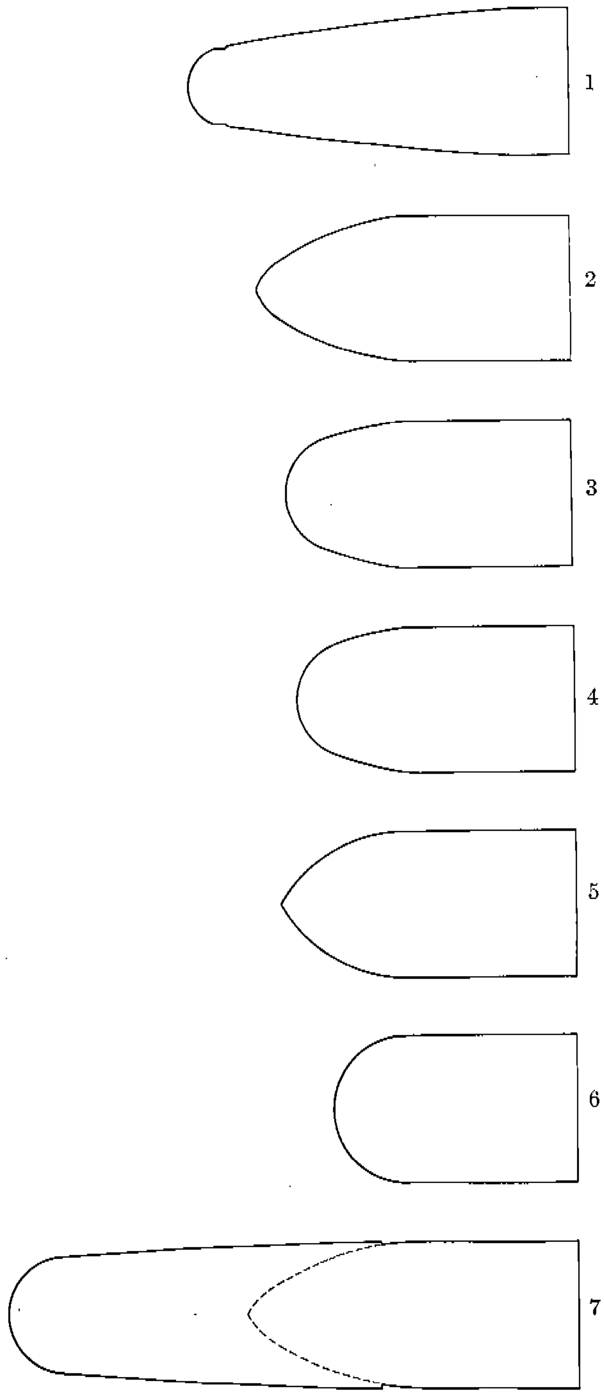

FIGURE 12. HVAR head shapes tested for underground trajectory.

CIT tests of ground penetration of the 5.0-in. HVAR are discussed only in the weekly progress reports.[37–39] The head shapes tested are shown in Figure 12 and the results are summarized as follows:

[redacted]

FIGURE 13. 11.75-in. sphere-ogive head after rocket had penetrated 75 ft of earth with impact velocity 1,275 fps. "Wart" on nose is fuse. Note that erosion extends only to intersection of sphere with 20-caliber portion (6½-in. diameter).

Type 1: Standard underwater sphere-ogive head. With this head, the rocket was entirely stable, traveling underground (in clay covered with sand) an average of nearly 50 ft, at dive angles of 15 to 20 degrees. The lateral deviation was very small, but deviations averaged several degrees from the mean, and, in contrast to underwater results, a considerable upward curvature of the trajectory was noted. The average round was recovered at a depth slightly less than half that corresponding to an extension of its air trajectory, and one, fired at 15-degree dive angle, actually emerged after 24 ft underground and detonated in the air (it carried a deceleration-discriminating fuze).

Type 2: Standard semi-armor-piercing Mk 2 head. All heads separated from their motors and emerged after 6 to 8 ft of travel underground, showing erosion on the nose and one side.

Types 3 and 4: Modifications of Type 2. Performance identical with Type 2. Apparently a portion of the ogive back of the spherical nose remained

FIGURE 14. Special 11.75-in. AP head after rocket had penetrated 65 ft of earth with impact velocity 1,225 fps. Note that erosion extends to full 11.75 in. diameter.

in contact with the "bubble," giving a large cross force.

Type 5: Blunt ogive head. Performance identical with Type 2.

Type 6: Hemispherical-nose head. All heads broke off from their motors, but the erosion was more symmetrical than in the unstable cases previously mentioned, indicating that drag rather than cross force may have been the primary factor in the failure.

Type 7: Sphere-cone ogive. This head appeared to be near the limit of stability since, although all heads broke off, they penetrated 12 to 14 ft and eroded quite symmetrically. Apparently the drag with this size of spherical nose is still too great.

Earth penetration tests with Tiny Tim have been made by NOTS, Inyokern, and one must consult Navy reports for the details. One such test gave the following results at impact angles of approximately 33 degrees:

Sphere-ogive head (Figure 13). This penetrated 70 ft in the same direction as the air trajectory for 1,275-fps striking velocity.

Mk 1 head. For striking velocity of 1,380 fps, penetration averaged 50 ft and the rounds turned up 10 degrees from their air trajectory. (At shallower angles one round broke.)

Special heavy head (Figure 14) having same exterior contour as Mk 1 but a greater length and weight. These rounds weighed approximately 1,550 lb instead of 1,120 as for the previous types. For an entrance velocity of 1,225 fps, their penetration characteristics were identical with those of the sphere-ogive heads. Why these should not turn up as the Mk 1 heads do has not been explained.

When heads having the same shape as the Mk 1 were fired with Mk 2 motors (wall thickness 0.240 in. instead of 0.300 in.), all motors were shattered, although their underwater performance is entirely satisfactory.

[redacted]

Chapter 25

EXTERIOR BALLISTICS OF SPIN-STABILIZED ROCKETS

By *C. W. Snyder*

25.1 SIMPLEST TYPE OF SPINNER MOTION: NUTATION

The motion of a spin-stabilized rocket in the absence of gravitational and aerodynamic forces is closely analogous to that of a finner in air. For the latter, the equilibrium position is one of zero yaw, and if displaced from it the rocket oscillates (in the plane determined by its axis and the tangent to the trajectory) with a frequency which increases with velocity at just the proper rate so that the distance traveled in each oscillation is a constant, σ. The equilibrium position of a spinner in the absence of air is also one of zero yaw. When displaced from this orientation, it oscillates so that the distance traversed during each oscillation cycle is a constant, λ, analogous to σ. Unlike the finner, however, the oscillations are not in one plane—the nose of the rocket moves in a spiral about the trajectory of the center of mass. This motion is called nutation; its projection on a plane through the trajectory duplicates exactly the oscillation curve of a finner. The constancy of the distance covered in each nutation cycle is a consequence of the fact that the rate of nutation is proportional to the rate of spin, which is, as indicated in Chapter 21, proportional during burning to the velocity. The analogy between finner and spinner motion is exact both during and after burning if one assumes that there is no jet malalignment, no aerodynamic forces on the spinner, and no damping forces on the finner.

Although these features of similarity between spinner and finner behavior are helpful, both the force system and the motion of spinners under conditions of reality are, in general, somewhat more complicated than those of finners. The complications result from the larger number of forces and moments which act on spinners, in combination with gyroscopic action.

25.2 FORCE SYSTEM OF SPINNERS

As was done for finners in Chapter 24, the first step is to set up a system of a small number of forces and torques which will be equivalent in effects to the multiplicity of distributed forces, both internal and external, which govern the motion of spinners. A detailed discussion of such a force system is given in *Exterior Ballistics*.[1]

The important elements of the system are five forces and four moments, as tabulated below.

The forces are

1. *Gravity*.
2. *Jet forces*, which act only during burning.
3. The *drag*, which, like that for finners, results from high air pressure on the nose, reduced pressure behind the rocket and skin friction.
4. The *lift* or *cross-wind force*, which accompanies yaw and causes planing action, tending to push the rocket in the direction of its yaw.
5. The *Magnus force*, an aerodynamic force peculiar to spinning projectiles.[a] It appears whenever there is a component of airflow perpendicular to the spin axis (i.e., when the yaw is not zero) and tends to move the rocket in a direction perpendicular to both the yaw and the trajectory. We can visualize it most easily if we consider the case where the rocket is oriented broadside to the relative wind (yaw = 90 degrees). The skin friction carries a certain amount of air around with the rocket as it rotates, and on one side of the rocket this trapped air collides with the air flowing past, creating a higher pressure, while on the other side the trapped air and the free air flow in the same direction giving reduced pressure. Theoretical analysis shows that the Magnus force is proportional to the product of the rocket's angular velocity by its linear velocity, and for smaller yaws than 90 degrees the factor $\sin \delta$ is also included.

The most important moments are:

1. The *overturning moment*, which tends to turn the rocket across the trajectory because the center of pressure (where lift and drag are assumed to act) lies forward of the center of mass. Finners have a righting moment instead.
2. The *Magnus moment* exists whenever the

[a] This is the force which causes a properly thrown baseball to curve.

Magnus force is applied elsewhere than at the center of mass. It is small in magnitude but important in effect.

3. The *spin deceleration moment*, which tends to slow down the spin because of air friction.

4. The *damping moment*, which always opposes the yaw, exists only when the yaw is changing and tends gradually to damp it out. It results from the difference in the forces on the two ends of the rockets associated with their different air velocities when the yaw is changing.

The greater complexity of these forces and moments as compared with those which act on a fin-stabilized projectile is apparent. For a finner, force 5 and moments 2 and 3 are entirely absent, while force 4 averages to zero because the rocket has a zero yaw on the average. If it is assumed that the overturning moment is proportional to the yaw angle (as was done also for finners and is approximately true for small yaws), then the equations of motion are linear, and the effects of the various forces may be computed separately and added to give the final motion. We shall confine ourselves mainly to this approximation since it will explain adequately the main features of spinner motion. There remain, however, a few important effects that require more complicated analysis.

The general features of spinner motion were sketched in Chapter 21, and it is suggested that the reader glance through the pertinent sections there before proceeding further. In the following paragraphs, we shall extend the analysis of Chapter 21, but without discussing the equations of motion from which the results are calculated. For further details the reader is referred to *Exterior Ballistics*[1] or to the original papers

It will clarify the following to keep in mind a particular rocket, and the 5.0-in./5 HCSR Model 34 (5.0-in. Rocket Mk 10 Mod 0) will serve as an example. It is described in Chapter 20. In Table 1 are given the pertinent ballistic constants for such a rocket. Slight changes from the actual constants have been made for convenience in applying the graphs to follow. Notation used in this chapter is the same as in Chapter 21, with certain additions, and is summarized in Table 2.

25.3 MOTION DURING BURNING

As indicated in Section 25.1, a spinner, in the absence of gravity and aerodynamic forces, will move along a straight trajectory with its nose oscillating in a spiral of constant nutation distance. In a real rocket, of course, this motion is modified. During the period of propulsion (burning period) the principal factors affecting the motion are the overturning moment, gravity, interaction with the launcher, and wind.

TABLE 1. Ballistic constants of typical 5.0-in. spinner.*

Stability factor during burning: $S = 2$

Radii of gyration:	$K^2 = 0.60 \text{ ft}^2$
	$k^2 = 0.030 \text{ ft}^2$
	$\frac{K}{k} = \sqrt{20} \approx 4.5$
Feet per turn:	$\nu = 6 \text{ ft}$
Feet per nutation:	$\lambda = 120 \text{ ft}$
Burning distance:	$d_b = 325 \text{ ft}$

Velocity parameter for the end of burning:

$$\zeta_b = \sqrt{\frac{d_b}{\lambda}} = \sqrt{\frac{325}{120}} = 1.50$$

Acceleration at 70 F: $G = 30g = 966 \text{ ft/sec}^2$

$$t_\lambda = \sqrt{2\lambda/G} = \sqrt{\frac{240}{966}} = 0.50 \text{ sec}$$

$$V_\lambda = \sqrt{2G\lambda} = \sqrt{240 \times 966} = 481 \text{ ft/sec}$$

$$\frac{1}{V_\lambda} = 0.00218 \text{ sec/ft.}$$

* The constants tabulated are approximately those of the 5.0-in./5 HCSR Model 34 which has an overall length of 32 in. (including nose fuze), a weight of 50 lb, and a velocity of 790 fps, and spins at 130 rps.

25.3.1 Effect of Overturning Moment

The overturning moment, the principal aerodynamic effect, introduces gyroscopic precession. Any uniform torque on a spinning gyroscope causes its axis to precess so that the motion of any point on the axis is a circle. The overturning moment acting on a spinner with a given yaw leaves the magnitude of the yaw constant but rotates the plane of yaw uniformly about the trajectory. In general, the initial conditions are not such as to give this dynamically stable mode of motion, but the nutations will be superimposed on it.

In the following discussion we shall frequently find it convenient to represent spinner motion

[redacted]

TABLE 2. Notation for spin-stabilized rockets.

$\zeta \equiv$ velocity parameter. $\zeta = \sqrt{d/\lambda}$; $\zeta_p = \sqrt{p/\lambda}$.
$\Theta \equiv$ characteristic function for a trajectory orientation (see Table 3).
$\theta \equiv$ orientation of the tangent to the trajectory relative to the launcher.
$\theta_0 \equiv$ quadrant elevation of the launcher.
$\lambda \equiv$ distance traveled in one nutation, assuming constant S (ft).
$\nu \equiv$ distance traveled in one rotation.
$\Phi \equiv$ characteristic function for orientation of rocket axis (see Table 3).
$\phi \equiv$ orientation of the rocket axis relative to the launcher.
$b \equiv$ subscript denoting "at the end of burning."
$d \equiv$ distance along trajectory from point of ignition (ft). $d = \frac{1}{2}Gt$.
$E \equiv$ function giving variation of malalignment effect with launcher length.
$G \equiv$ acceleration of the rocket in horizontal fire (ft/sec^2).
$g \equiv$ acceleration of gravity (32.2 ft/sec^2).
$K \equiv$ transverse radius of gyration (ft).
$k \equiv$ polar radius of gyration (ft).
$l \equiv$ length of rocket (ft).
$p \equiv$ launcher length (ft). As a subscript, it signifies "at the end of the launcher."
$q \equiv$ transverse angular velocity of mallaunching (radians per second). As a subscript, it denotes "produced by mallaunching."
$R_m \equiv$ jet malalignment (ft).
$S \equiv$ stability factor [see equation (23) of Chapter 21].
$s \equiv$ spin angular velocity (radians per second).
$t \equiv$ time (seconds).
$t_\lambda \equiv$ time required to complete first nutation (assuming nutation and acceleration to commence simultaneously and rocket to continue burning throughout the nutation). $t_\lambda = \sqrt{2\lambda/G}$.
$u \equiv$ unbalance. Subscripts S and D denote static and dynamic unbalance.
$v \equiv$ velocity (fps).
$v_\lambda \equiv$ velocity at the end of the first nutation (same assumptions as for t_λ). $V_\lambda = \sqrt{2G\lambda}$.
$W \equiv$ wind velocity (fps). As a subscript, it denotes "produced by wind."
$W_X \equiv$ wind velocity component perpendicular to launcher (fps).
$X, Y \equiv$ coordinates in a plane perpendicular to the launcher. X is positive to the right and Y is positive down.

graphically by using a moving system of coordinates having its origin at the center of mass of the rocket, its Z axis pointing in the direction of the launcher, its X axis pointing to the right, and its Y axis pointing down. The change of the rocket from its original orientation (the Z axis) can then be represented by the projection on the XY plane of a point 1 ft ahead of the center of mass, and, in the approximation of small angles, the distance of the projected point from the origin is proportional to the orientation angle. As the motion proceeds, this point will trace out a curve which is easily interpreted by imagining one's self standing behind the rocket and watching the motion of the nose. Such curves we shall call "orientation curves," and a number of them will be included later in this chapter. A much more complete set is contained in *Exterior Ballistics*.[1]

As in the previous chapter, we shall use three angles to specify the rocket's position and motion:

$\theta \equiv$ angle between the launcher line and the tangent to the trajectory,
$\phi \equiv$ angle between the launcher line and the rocket axis, and
$\delta \equiv$ angle between the rocket axis and the tangent to the trajectory.[b]

Since the motion is not plane, we shall have to give the projections of these angles on the horizontal and vertical planes, and shall denote the projections by subscripts X and Y, respectively.

The orientation curve for a precession or an undamped nutation is a circle, and it is simple to superimpose the two circular motions provided that we know their relative velocities. From an analysis which includes the effect of the overturning moment, but excludes other aerodynamic forces and gravity (which would introduce only minor corrections), we find

Angular velocity of nutation =

$$\frac{sk^2}{2K^2}(1 + \sqrt{1 - 1/S});$$

Angular velocity of precession =

$$\frac{sk^2}{2K^2}(1 - \sqrt{1 - 1/S}).$$

From the ratio of these we find that the number of nutations for each precession is

1.00 for $S = 1.00$ (very low stability factor);
5.82 for $S = 2.00$;
9.86 for $S = 3.00$;
$4S - 2$ as the limit approached for very large S.

Thus the distinction between nutations and precessions virtually disappears for very low stability factors.

At the same time, of course, the rocket is rotating (spinning) about its oscillating axis with a higher angular velocity s. Dividing this by the angular

[b] Evidently in order to draw an orientation curve for the yaw angle δ, we should have to take the Z axis pointed along the trajectory instead of along the launcher, but only one such curve is given in this book.

velocity of nutation, we get the number of spin rotations per nutation as

$$2.0 \frac{K^2}{k^2} \text{ for } S = 1.0;$$

$$1.17 \frac{K^2}{k^2} \text{ for } S = 2.0;$$

$$1.10 \frac{K^2}{k^2} \text{ for } S = 3.0;$$

$$1.00 \frac{K^2}{k^2} \text{ as the limit approached for large } S.$$

Since ν is the distance traveled during each rotation, the distance for each nutation is, for large values of S,

$$\lambda = \frac{\nu K^2}{k^2}.$$

This depends only on geometrical constants of the rocket. For values of S customarily used for ground-fired spinners, this expression gives a result about 15 per cent lower than that observed.

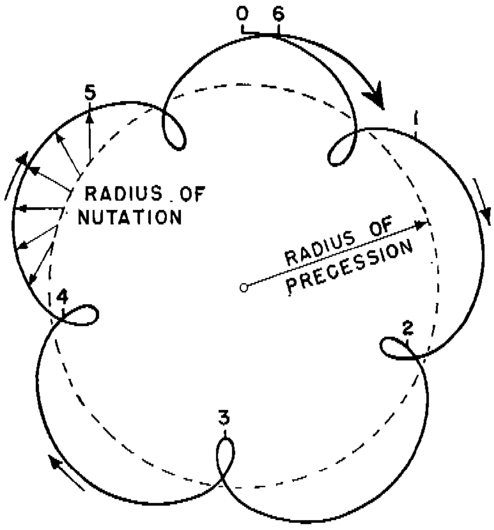

FIGURE 1. Precession and nutation without damping ($S = 2$).

The orientation of a precessing and nutating rocket with a stability factor of 2 is shown graphically in Figures 1 and 2. The first shows the case where the nutation amplitude is constant and one-fourth that of the precession, and the second shows a case of extremely large damping where the amplitude of the nutation decreases to 0.7 times its former value during each nutation and where the rocket is released with zero yaw, so that initially the nutation and precession amplitudes are equal. The numbers along the curves indicate the ends of each nutation.

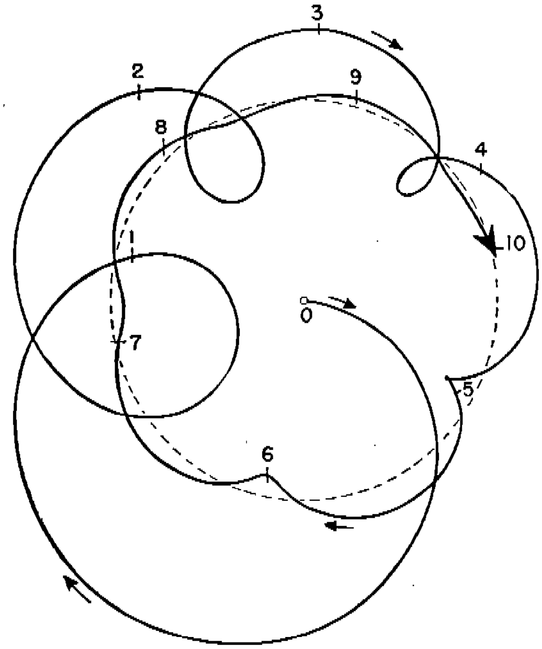

FIGURE 2. Precession and damped nutation ($S = 2$).

25.3.2 Effect of Gravity

If no aerodynamic forces were acting, the effect of gravity would be simply a vertical drop of the trajectory. Thus our hypothetical HCSR fired horizontally from a zero-length launcher would have an acceleration g downward and $30g$ forward so that its center of mass would move in a straight line falling below the horizontal by an angle whose tangent is 1/30, i.e., by 33 mils. Since its nose would continue to point in the direction of launching, it would have a 33-mil yaw upward. After burning, it would of course move in a parabola instead of a straight line.

In the presence of the overturning moment, the up yaw caused by the gravity drop lifts the nose, inducing a precession first to the right and then down. The process is slow because the magnitude of the yaw causing it starts at zero and builds up slowly, but by the end of burning the rocket will be

[redacted]

somewhat to the right of the launcher line and, if the burning time is long enough, may drop well below the point where gravity alone would have taken it. This effect is calculated in reference 2 and shown graphically in Figures 3 and 4, which give the orientation curves for the rocket axis and the trajectory, respectively. As in Figure 4 of Chapter 24, the quantities shown in the graphs are "characteristic functions" Θ and Φ. To obtain the

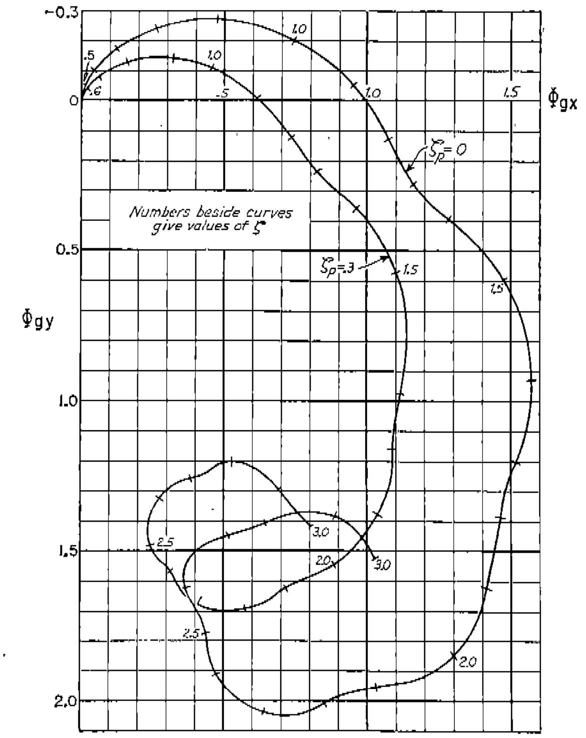

FIGURE 3. Deflection of the rocket axis due to gravity, during burning ($S = 2$).

actual angles θ and ϕ in radians for any particular rocket, the functions must be multiplied by the factor g/G for horizontal launching, or in general for a quadrant angle θ_0, by the factor $g \sin \theta_0 / (G - g \sin \theta_0)$.[c] Thus, in the particular case we are considering, the point at the end of burning ($\zeta = 1.50$) corresponds to

$$\Phi_{gX} = 1.47; \quad \Phi_{gY} = 0.6;$$

$$\Theta_{gX} = 0.56; \quad \Theta_{gY} = 1.05.$$

for the zero-length launcher. The conversion factor $g/G = 1/30$ so we calculate that the rocket is pointing 49 mils to the right and 20 mils below the launcher, and the trajectory is deflected 18.7 mils to the right and 35 mils downward. Here the downward deflection is barely greater than it would be in the absence of the overturning moment, but it is apparent from the curves that with a little longer burning time it would become much greater.

[c] Relations between characteristic functions and actual angles are given in Table 3 for all functions used in this chapter.

TABLE 3. Relations for converting from characteristic functions to actual angles.*

Gravity:

$$\theta_g = \frac{g}{G}\Theta_g.$$

Mallaunching:

$$\theta_q = q t_\lambda \Theta_q.$$

Wind:

$$\theta_W = \frac{W_N}{v_\lambda}\Theta_W.$$

Relations between ϕ and Φ are identical.

$$t_\lambda = \sqrt{2\lambda/G}$$

$$v_\lambda = \sqrt{2G\lambda}$$

$$\zeta = \sqrt{d/\lambda}$$

$$\zeta_p = \sqrt{p/\lambda}$$

* All above relations assume horizontal fire. If quadrant elevation is θ_0 substitute $G - g \sin \theta_0$ for G and $g \cos \theta_0$ for g wherever they appear.

Each of the curves of Figure 4 shows a minimum of right deflection for $\zeta \approx 2.8$, because slightly before this the rocket has made one complete precession and is ready to start heading off toward the right again. For higher stability factors, the rocket travels farther in one precession, and the gravity deflections for a given burning distance are somewhat less.

Curves giving the deflection of the center of mass from the range line throughout burning are also given in reference 2, but in most actual cases where the total flight distance is considerably greater than the burning distance, this deflection may be neglected, and the trajectory angle at the end of burning will give the final deflection with sufficient accuracy except for drift effects.

After burning ceases, the curves of Figures 3 and 4 are no longer applicable; the rocket tends to settle into the position where the yaw to the right produces enough precession to cause it to follow the trajectory, as explained in Chapter 21.

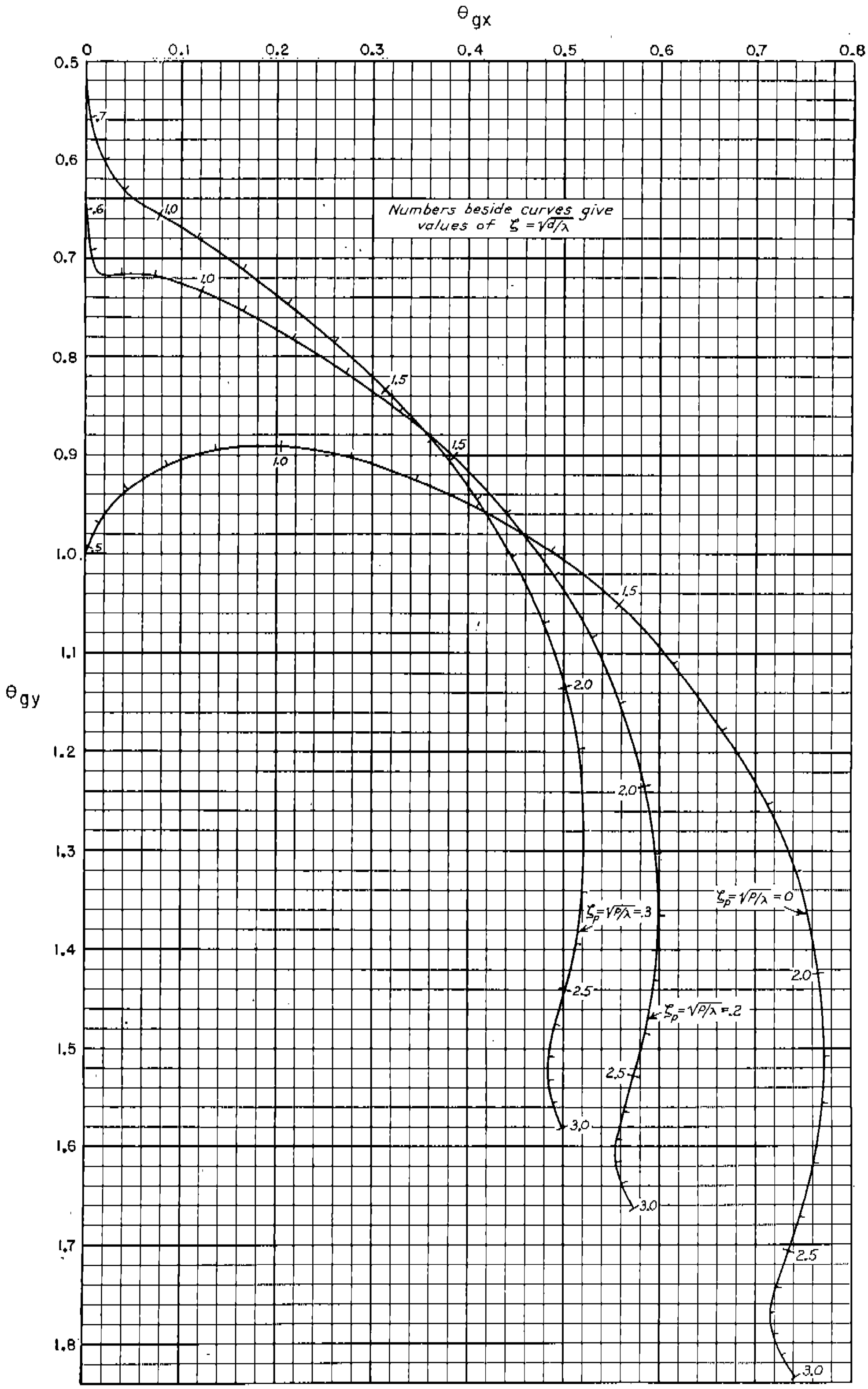

FIGURE 4. Deviation of the trajectory due to gravity, during burning ($S = 2$).

[redacted]

25.3.3 Effect of Mallaunching

One of the most important factors in spinner motion, and the most difficult to control, is mallaunching. The term "mallaunching" is used technically to denote any angular velocity, about a transverse axis, which the rocket acquires during launching. Such angular velocities may be produced by gravity (tip-off), a faulty launcher, dynamic and static unbalance of the round, elliptical bourrelets, or jet malalignment.

Because the effect of mallaunching in deviating the trajectory occurs almost entirely in the early part of burning before the velocity and the aerodynamic forces become large, a fairly satisfactory treatment of it can be obtained by assuming that no aerodynamic forces act on the rocket. References 3 and 4 contain this analysis. The more general case where the effect of the overturning moment is included is discussed in reference 2, and both cases are treated in *Exterior Ballistics*.[1]

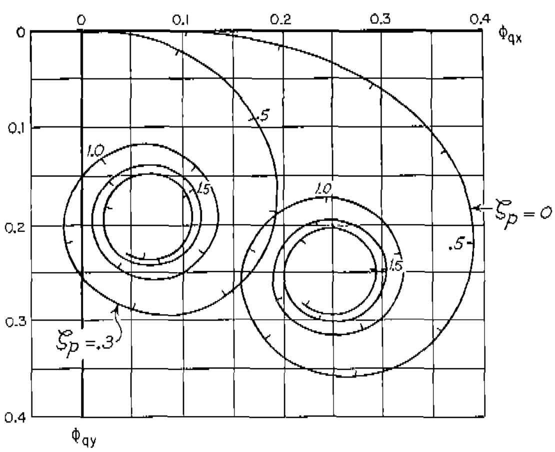

FIGURE 5. Deflection of the rocket axis due to mallaunching, during burning ($S = 2$).

If one assumes that the launcher is absolutely rigid and that there is no friction, malalignment, or unbalance, the angular velocity produced by tip-off is computed easily by considering the gravity torque acting on the rocket, supported on its rear bourrelet, during a time equal to that between the arrival of the front and rear bourrelets at the end of the launcher. The resulting equations are given in reference 4 and are identical with those for finners because the gyroscopic forces can produce no significant effect in so short a time. Practical launchers are not absolutely rigid, and their reaction on the round may impart to it either more or less angular velocity than the simple theory would predict. It is this variation in mallaunching that produces the sometimes rather large discrepancies in centers of impact among different launchers.

If, on leaving the launcher, a rocket receives an angular velocity throwing the nose downward, for example, it responds in the manner that we have by now come to expect, changing the downward motion into motion to the left. Here, however, we have to do, not with a precession, which is the response to the continued action of a force, but with a nutation, which is roughly $4S$ times more rapid than a precession. The nose moves in a tightening spiral[d] so that virtually all the change in orientation occurs in the first nutation, as shown by Figure 5 in terms of symbols similar to those of Figure 3, except that we must rotate the figure clockwise 90 degrees in order to apply to tip-off. To get the actual angles, we multiply the tabulated functions by the factor $t_\lambda = \sqrt{2\lambda/G}$;[e] the result is expressed in angular units per unit of mallaunching velocity. For our hypothetical HCSR, the factor is 0.50 for horizontal fire.

Using the curve for the zero-length launcher, we find that by the end of burning ($\zeta = 1.5$) the rocket has completed $2\frac{1}{4}$ loops on its spiral and has coordinates

$$\Phi_{qX} = 0.3; \quad \Phi_{qY} = 0.25;$$

corresponding to an orientation 0.15 degree (or mil) below and 0.125 degree (or mil) left of the launcher line for an initial angular velocity of 1 degree (or mil) per second.

After the end of burning, in the absence of aerodynamic forces, the nose would move in a circle having the same center and radius as the spiral had when the thrust ceased. With the overturning moment acting, this nutation will, of course, be superimposed on the precession.

The direction of the trajectory during burning is given similarly in Figure 6.[f] Again we turn the

[d] The reader may recognize it as a Cornu spiral, which gives another representation of the Fresnel integrals which appear so frequently in the theory of both finner and spinner trajectories.

[e] As before, we use $G - g \sin \theta_0$ in place of G if the quadrant elevation is greater than zero.

[f] Figure 5 represents the vacuum case, but Figure 6 includes the aerodynamic overturning moment, the effect of which is quite small in this case.

[redacted]

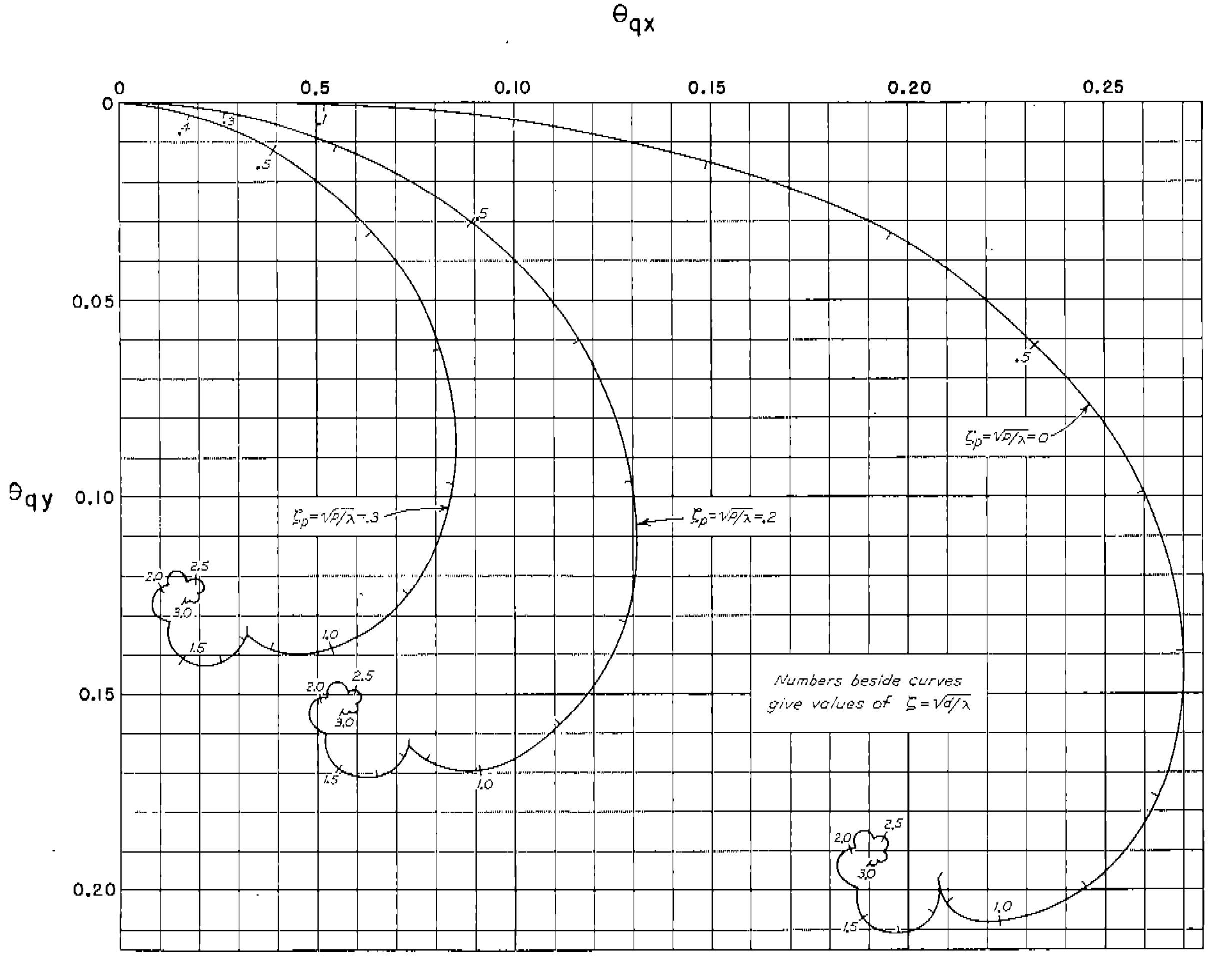

FIGURE 6. Deviation of the trajectory due to mallaunching, during burning ($S = 2$).

figure through 90 degrees to apply to tip-off and examine the case $\zeta_p = 0$, obtaining

$$\Theta_{qX} = 0.188; \quad \Theta_{qY} = 0.207.$$

The conversion factor is again 0.50, so that the trajectory angles are 0.094 mil down and 0.104 mil left for each mil per second of initial angular velocity. Thus it requires a tip-off of 180 mils per second to more than offset the approximately 19 mils right deflection which we calculated for the gravity effect. In practice, longer launchers are used, reducing the gravity effect relative to the tip-off effect, and the tip-off is large enough (approximately 100 mils per second for the 5.0-in. GPSR[g] and 2 or 3 times this for some rockets), so that it usually predominates, and the rocket has a left orientation throughout burning and drifts steadily to the left.[h]

In discussing dispersion we shall be interested in the magnitude of the trajectory deflection without regard to direction for various launcher lengths. Measuring the radii from the origin to the $\zeta = 1.5$ points on the three curves of Figure 6, we obtain

$$\zeta_p = 0; \quad p \approx 0; \quad \Theta = 0.278; \quad \theta/q = 0.139;$$

$$\zeta_p = 0.2; \quad p \approx 5 \text{ ft}; \quad \Theta = 0.178; \quad \theta/q = 0.089;$$

$$\zeta_p = 0.3; \quad p \approx 11 \text{ ft}; \quad \Theta = 0.142; \quad \theta/q = 0.071;$$

where θ/q is the actual trajectory angle at the end of burning for unit mallaunching.

[g] Described in Chapter 20.

[h] The calculated orientation at the end of burning, analyzed into gravity and tip-off effects are tabulated for several 5.0-in. spinners in reference 5.

[redacted]

25.3.4 Wind Effect

One more factor of importance during burning is the wind, which may alter the trajectory significantly. For the component of wind along the range line, the effect is nonlinear and quite complicated,[6] so we shall discuss only the cross-wind effect. If a wind is blowing across the launcher from left to right, the effect is essentially as if the rocket were launched into still air with a yaw to the right. Hence an overturning moment exists because of the wind, and the rocket precesses clockwise as would be expected. Nutation is of little importance in this motion, and the deflection is slow and spread out through the whole of burning instead of taking place mostly in the first nutation as in the case of mallaunching. Also in contrast to the mallaunching effect, it is relatively insensitive to launcher length.

The characteristic functions for cross wind are plotted in Figures 7 and 8. The conversion factor in this case is $1/v_\lambda = 1/\sqrt{2G\lambda}$, the results applying to unit wind velocity. It will be noted that a positive wind increases the gravity drop in all cases, but the lateral deflection starts downwind and then reverses if the burning continues long enough. After burning, the deflection is naturally downwind because of the downwind component of the drag just as in the case of finners.

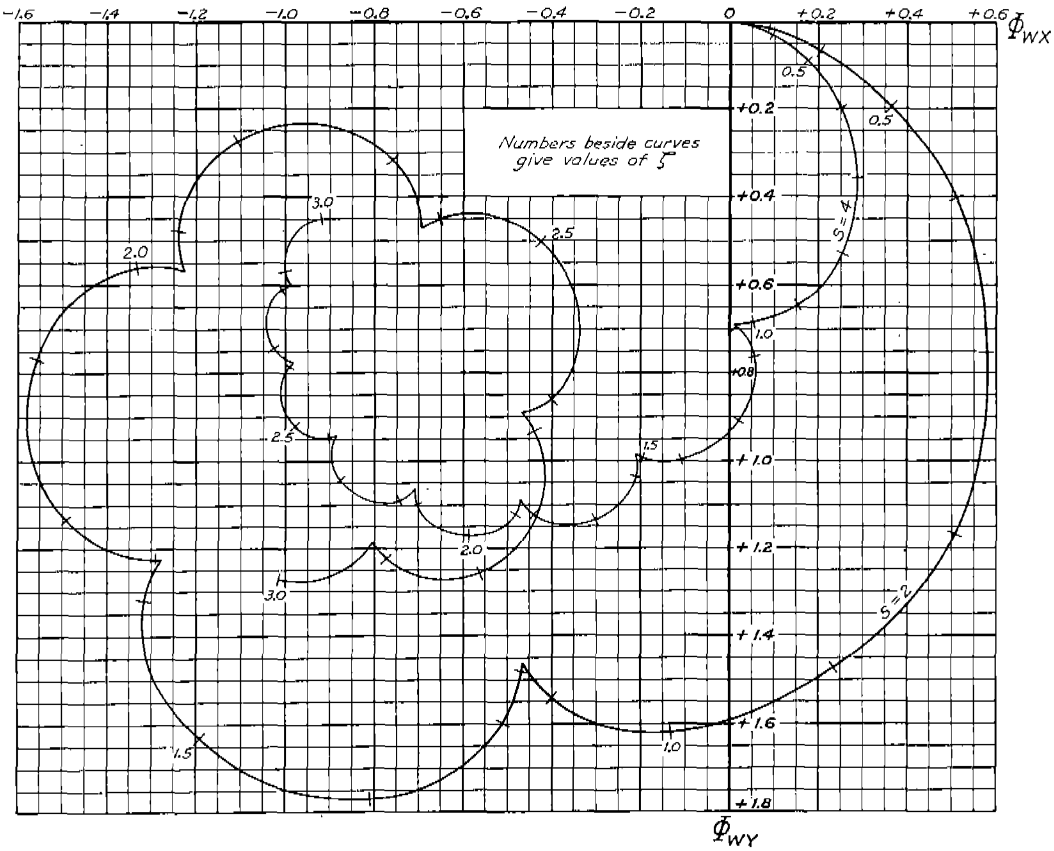

FIGURE 7. Deflection of the rocket axis due to cross wind during burning ($S = 2$).

Numerical values for our hypothetical example are (using $\zeta_p = 0$)

$$\Phi_{WX} = 1.19; \qquad \Phi_{WY} = 1.63;$$
$$\Theta_{WX} = -0.03; \qquad \Theta_{WY} = 0.87.$$

Using the conversion factor 2.18×10^{-3}, we find that at the end of burning a cross wind toward the right of 1 fps will tip the rocket axis 2.6 mils left and 3.6 mils down, and deviate the trajectory 0.065 mil left and 1.9 mils down. The very small value of the lateral deviation is obviously an accidental result of the particular burning distance chosen. Increasing or decreasing the burning distance by a factor of 2 would increase the deviation more than tenfold.

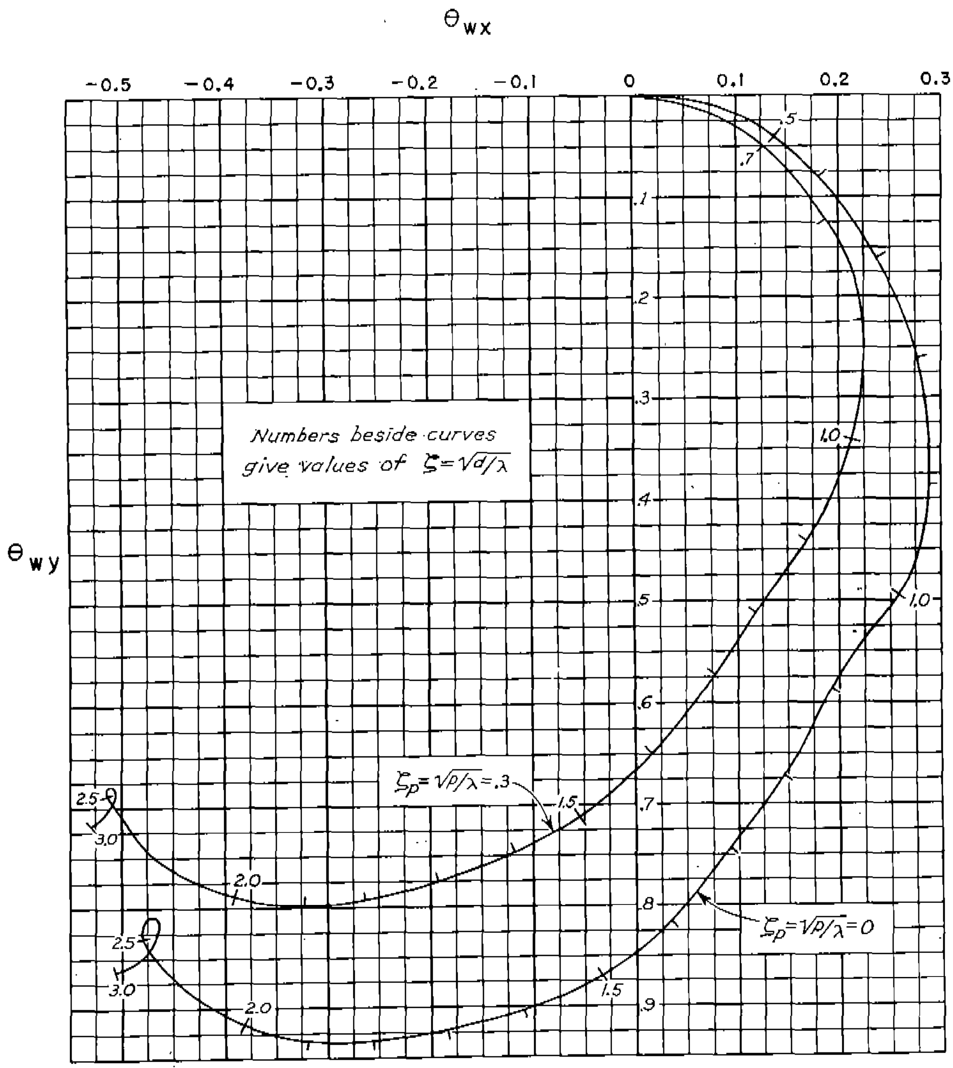

FIGURE 8. Deviation of the trajectory due to cross wind during burning ($S = 2$).

The calculations do show, however, that wind sensitivities of 1.5 to 2 mils per fps are obtained for low-stability spinners, so that gusty winds varying in velocity by only 5 or 10 fps can easily double the dispersion. As the stability factor increases, the characteristic curves for cross-wind effect hug the vertical axis more and more closely, and the vertical deflections also decrease, so that, if the dispersion produced by variable cross wind is to be kept low, a high stability factor is essential.

25.4 MOTION AFTER BURNING

In all of the foregoing discussion of motion during burning, the only aerodynamic effect which has been assumed to be acting is the overturning moment. This is permissible because all other aerodynamic effects are smaller and do not make themselves felt because of the short time involved. After burning, the times involved are, in general, considerably longer and virtually all the forces and torques may have observable consequences.

25.4.1 Gravity

As has already been mentioned, the primary effect of gravity is in producing curvature of the trajectory so that the rocket assumes an equilibrium yaw to the right. In addition, for high-angle fire, it causes large changes in velocity so that the aero-

[redacted]

dynamic forces, and hence also the stability factor, vary widely in different portions of the trajectory.

25.4.2 Drag

The drag force reduces the velocity gradually without affecting the spin. In the absence of other factors, it would gradually increase the stability, but the much larger changes in velocity caused by gravity make its effect relatively insignificant.

25.4.3 Lift and Magnus Force

The equilibrium yaw to the right after burning causes the cross-wind force to be directed toward the right and the Magnus force to be directed upwards,[i] and both forces produce drifts. The theoretical treatment of these effects is not very satisfactory, and the reader is referred to *Exterior Ballistics*[1] for quantitative details. We would expect, however, that the drift to the right would be approximately proportional to the equilibrium yaw angle and hence [from equation (24) of Chapter 21] proportional to the angular velocity of spin for a given quadrant angle. It is proportional also to the flight time and to the angle of elevation.[7] For the 5.0-in./5 HCSR fired at 45-degree elevation, the drift amounts to approximately 34 mils.

The Magnus force is proportional to the spin velocity and to the broadside area (hence, for a given caliber, proportional to the length L), so that the soaring effect, which increases the range, should be proportional to the factor s^2L. This soaring effect is difficult to separate from other effects, but it appears to increase the maximum range of the 3.5-in. spinner by about 5 or 10 per cent.

25.4.4 Spin Deceleration Moment

The spin deceleration moment, in addition to its obvious role of reducing the spin, tends slightly to increase the amplitude of the nutations. This can be understood by noticing that its effect is directly opposite to that of the jet force in accelerating the spin during burning, so that it tends to move the rocket outward along the Cornu spiral of Figure 3. The effect is small, but not insignificant, for we shall see that damping the nutations is all-important in achieving stability in high-angle fire.

[i] This is exactly true only if the Magnus moment is zero.

25.4.5 Damping Moment

In analogy with finners, the damping moment serves to remove energy from the nutations. In this role, however, it is overshadowed for spinners by the Magnus moment.

25.4.6 Magnus Moment

The most obvious effect of the Magnus moment is to alter the equilibrium yaw so that it is not directly to the right but is below or above this position according to whether the point of application of the Magnus moment is ahead of or behind the center of mass. Its most important role is in connection with stability, as discussed in the following section.

25.5 STABILITY

The term "stability" has a rather wide variety of meaning. As applied to spinning rockets, it usually means that the yaw is small during the whole flight and undergoes no sudden changes. Small yaw is necessary in order to keep the drag low, to avoid losses in range and striking velocity, in order to minimize dispersion, and in order to have the rocket strike nose first as required for proper fuze operation. In Chapter 21 we noted that one condition necessary for stability is that the gyroscopic forces, expressed by the stability factor S, be sufficiently large. If, for example, $S = 0.96$, the nutation amplitude is multiplied by 3.5 every nutation, or by 525 every five nutations, and the nose of the rocket is very soon traveling in a spiral of radius comparable with the length of the round with a yaw that may be 30 degrees or more. In practice the only feasible method of increasing stability is by increasing the spin. If the stability factor is high enough to get the rocket safely to the end of burning, no later trouble from this source will develop, since the drag reduces the velocity faster than the spin deceleration moment reduces the spin.

When a spinner is fired at an elevation angle too high for its rate of spin, an entirely different type of

[redacted]

instability sets in at or somewhat beyond the peak of the trajectory. The yaw builds up suddenly to a very large value, the rocket emits a noise which has come to be known among range workers as a "wow-wow," and the projectile strikes the ground approximately broadside and usually considerably to the left of its normal impact point. This behavior occurs because the gyroscopic stability prevents the rocket from aligning itself promptly with the rapidly changing direction of the trajectory, so that the yaw exceeds a certain critical value. What determines the critical yaw we shall see presently.

We have seen in Chapter 21 that a spinner is able to follow its curved trajectory because it has an equilibrium yaw to the right so that the overturning moment makes the nose precess downward. As indicated by equation (24) of Chapter 21, the magnitude of this equilibrium yaw for any point on the trajectory is proportional to the component of gravity normal to the trajectory and inversely proportional to the velocity. Both these factors vary in such a way as to make the equilibrium yaw a maximum at the peak of the trajectory and critically dependent on the quadrant elevation. As an example,[7a] a rocket which has an equilibrium yaw of 1 degree at the end of burning for horizontal fire may have the following values at the summits of high-angle trajectories:

	Degrees				
Angle of elevation	30	40	50	55	60
Equilibrium angle of yaw	2.30	3.6	6.4	9.4	14.5

These values are probably a fairly good approximation to the equilibrium yaws of the 3.5-in. spinner, but are too high for most of the 5.0-in. barrage spinners.

Our assumption that the overturning moment is proportional to the yaw or to the sine of the yaw is obviously false for large yaws. Long before the yaw becomes 90 degrees, this moment goes through a maximum and then usually decreases to zero and changes into a righting moment. A spinning rocket for which the overturning moment is negative will apparently have its equilibrium yaw to the left, and hence the earliest explanation of the "wow-wows" was as follows.[j]

As the projectile approaches the summit, the tangent to the trajectory turns more and more rapidly, and the projectile must yaw farther and farther right so that the aerodynamic moment will be large enough to cause the nose to precess downward at the same rate that the trajectory turns. Eventually it reaches the angle corresponding to maximum overturning moment, and for greater yaws the moment decreases; then it is impossible for the axis of the projectile to turn as rapidly as the trajectory does. The nose continues to precess down and to the right, but the trajectory turns downward much more rapidly so that the yaw increases to the point where the aerodynamic moment reverses sign

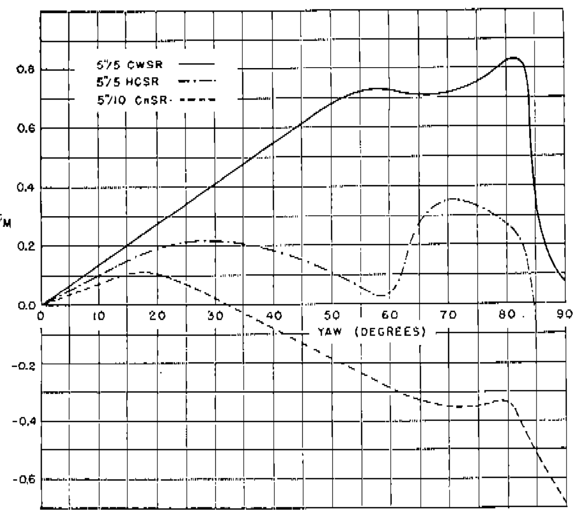

FIGURE 9. Variation of overturning moment coefficient with yaw for typical 5.0-in. spinners. C_M is proportional to the overturning moment divided by the square of the velocity.

and becomes a righting moment. As long as this moment is no larger than the maximum overturning moment, the nose of the rocket precesses upward and back to the left at a relatively slow rate; but if, as the velocity and yaw increase, the righting moment becomes large enough, there is a new equilibrium yaw position in which the rocket has a large left yaw. Its axis then spirals around this new equilibrium position with an amplitude that is very large because the initial position was so far from the equilibrium position.

This theory explained the qualitative behavior very well, but it broke down completely as soon as wind and water tunnel data and especially yaw camera data began to become available. Thus it was found that most spinners become unstable at an equilibrium yaw in the neighborhood of 10 degrees, whereas the yaw for which the overturning moment is a maximum is always considerably greater than this value. The actual variation of overturning

[j] This explanation is derived in greater detail in reference 7.

[redacted]

moment with yaw differs greatly for different rockets, as can be seen in Figure 9, where the quantity plotted is the overturning moment coefficient.[k]

The true nature of the instability was first revealed by yaw camera records [l] such as those in Figure 10. This record shows the variation over an interval of about 10 seconds in the angle between the axis of the rocket and the rays of the sun. The oscillations whose amplitude is increasing nearly exponentially are the nutations. The time scale is defined by the 0.14-second period of the nutations. Evidently the instability is due to a building up of the nutations rather than any change in the precessional motion. Records covering the early part of an unstable trajectory show that for several seconds after launching the nutations are damped in the same way that they are throughout all of a normal trajectory. Apparently, when the yaw exceeds a certain critical value, something begins to pour energy into the nutational motion. It was shown in reference 11, and in greater detail in *Exterior Ballistics*,[1] that the Magnus torque is the only aerodynamic force which, averaged over a nutation, can add or subtract a significant amount of energy, and that it is responsible for the instability.

The direction of the Magnus force is perpendicular to the trajectory and to the plane of yaw, and its point of application depends rather critically on the yaw. For very large yaws, it is probably at the center of figure of the rocket, which is usually slightly back of the center of mass, but for small yaws it is usually ahead of the center of mass. The Magnus *force* probably varies fairly closely with the sine of the angle of yaw, but, because of the shift in its point of application from ahead of to behind the center of mass, the Magnus *moment* is positive (i.e., overturning) for small yaws and negative for large yaws, and its maximum positive value may occur for yaws of only a few degrees. The damping effect of the Magnus moment is easily understood from Figure 11. In part A of the figure is plotted a hypothetical variation of Magnus moment with yaw, and in B we consider the magnitude and direction of the Magnus torque during a single nutation for two cases where the equilibrium yaws correspond to points A, B, and C.

To simplify the figure, the precessional motion is omitted, and as usual the rocket's varying orienta-

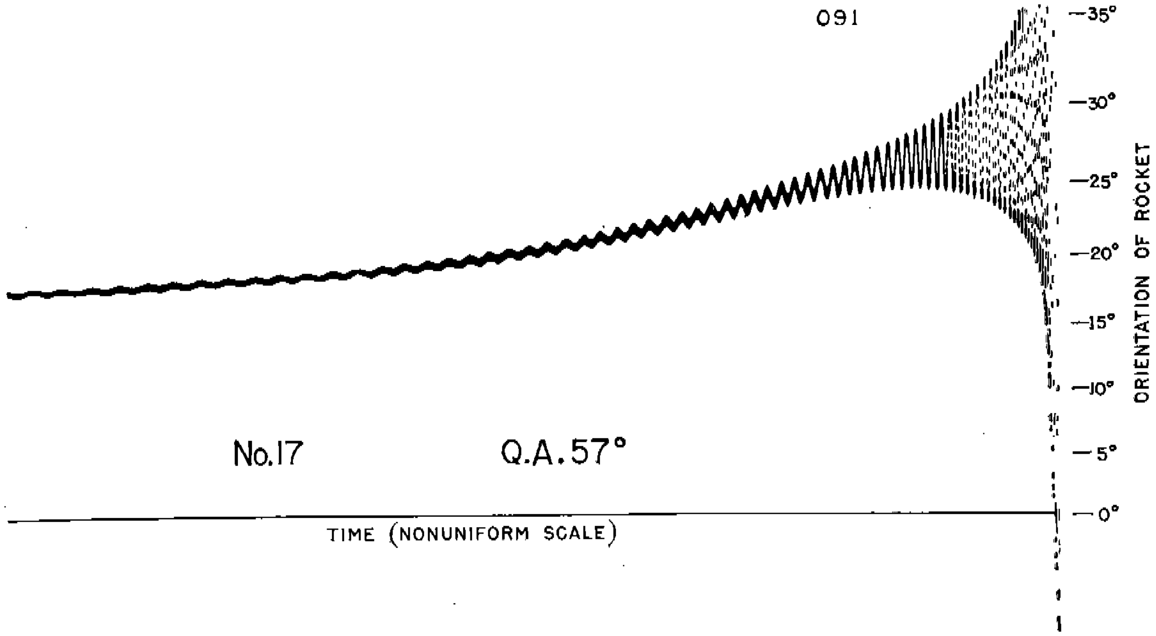

Figure 10. Yaw camera record for spinner which becomes unstable because of negative damping of the nutations at the peak of the trajectory.

[k] Curves are reproduced from a local memorandum[8] based on data from a National Bureau of Standards report[9] on wind tunnel measurements and on various reports on high-speed water tunnel measurements by the CIT Hydraulic Machinery Laboratory.

[l] The yaw camera is described in *Field Testing of Rockets*,[10] one of the CIT OEMsr-418 final reports.

tion is represented by the curve (a circle) traced by its nose as we look along the trajectory in the direction of motion, the difference between this and previous figures being that the Z axis is now along the trajectory instead of along the launcher. The straight arrows represent the torque at various points during the nutation, their magnitudes being obtained from the upper curve and their directions [m] being at right angles to the line representing the yaw (i.e., the line from T to the point on the circle). It is evident that for small yaws (points A and C) the net effect of the torques is to oppose the motion and hence damp out the nutations, whereas for large yaws (point B), the net effect is in the same direction as the motion, and the damping is negative.

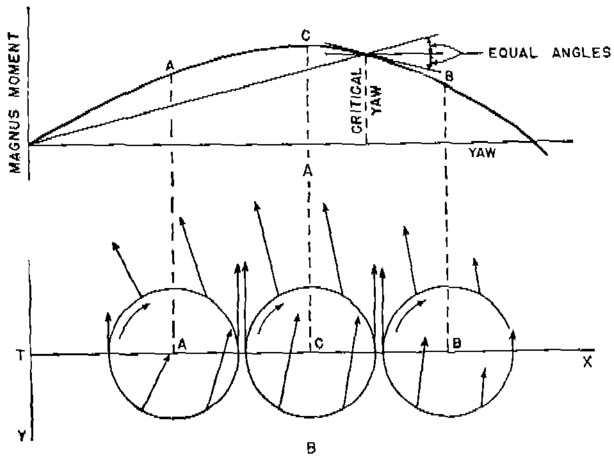

FIGURE 11. Diagram illustrating the effect of the Magnus moment during one nutation.

The derivation of the exact critical yaw which separates positive from negative damping is somewhat involved, but the result is shown in part A of the figure. One might expect it to be at the exact peak C, but it is displaced slightly beyond by the fact that the vectors representing the torques are not all parallel.

When, near the peak of the trajectory, the equilibrium yaw exceeds the critical yaw, the nutation amplitude begins to build up slowly so that it will be somewhat beyond the peak that the actual "wow-wows" begin. In fact, if the angle of elevation is only very slightly too large, the equilibrium yaw may decrease below the critical point on the descending part of the trajectory before the nutations have built up significantly, so that nothing noticeable happens even though the damping was negative for a time.

[m] These arrows are not conventional torque vectors but point in the direction which the torque tends to move the nose of the rocket.

25.5.1 Effect of Wind on Stability

From the preceding analysis we can immediately derive one important effect of down-range winds, the treatment of which we have omitted because of its complexity. Obviously a wind in the direction of the motion will reduce the aerodynamic forces, since they depend on the *relative* velocity between rocket and air; thus a larger yaw angle will be required to give enough precession to turn the rocket over the top of its trajectory, and it will become unstable at somewhat lower elevation angles. An up-range wind, on the other hand, increases the maximum angle of elevation at which stability over the trajectory peak can be retained.

25.6 DISPERSION OF SPINNERS

The two principal advantages of spinners over finners are their more convenient shape and their usually smaller dispersion. Their greater accuracy stems from the fact that the spin changes the direction of the malalignment torque so rapidly that it averages approximately to zero, thus by-passing the barrier of gas malalignment which limits the accuracy of finners. The introduction of spin, however, creates many more new problems than it solves, and considerable effort is required if the dispersion of a spinner is to be much less than half that of a typical well-designed finner.

Dispersion of spinners may arise from any of the following causes:

1. Variation in wind velocity.
2. Variation in tip-off.
3. Out-of-roundness of the bourrelets.
4. Static and/or dynamic unbalance.
5. Malalignment.

We have already treated the wind effect and have seen that it can be reduced by using longer launchers or by increasing the stability factor. Causes 2 and 3 lead to dispersions which are smaller than those caused by 4 and 5, and which depend on such variables as launcher length and burning time in the same way as the latter, so we shall not discuss them here. If the rounds have been carefully bal-

[redacted]

anced, the out-of-roundness might become important, however; it is discussed further in *Exterior Ballistics*.[1]

25.6.1 Unbalance

A symmetrical body is said to be statically unbalanced when its center of mass does not lie on its axis (see Figure 12). For spinners, we will take the

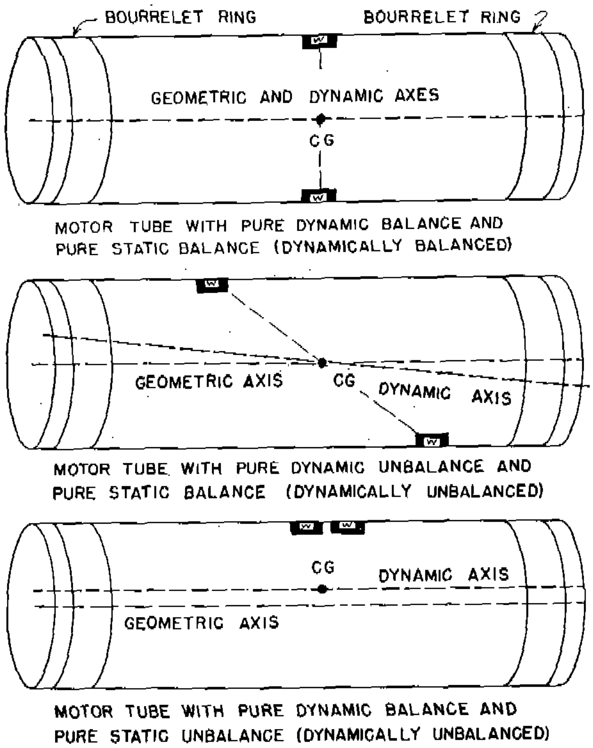

FIGURE 12. Types of unbalance of a spinner. The small weights W represent overweight sections of the tube such as occur from inequality in wall thickness. The center of gravity is the point CG.

axis to be that of the bourrelets since this is the one about which the rocket is forced to rotate in a rigid, snug-fitting launcher. As a quantitative measure of static unbalance we will take the angle u_S defined by the ratio of the distance between the center of mass and the bourrelet axis to the distance between the bourrelets.

In a spinner with this sort of unbalance,[n] the sides of both bourrelets toward which the center of mass is displaced will exert a (centrifugal) force on the launcher guides, which thus must exert a (centripetal) reaction force to maintain the rotation about the bourrelet axis. After the front bourrelet clears the launcher, it is no longer subject to this reaction force, but the rear bourrelet is. The result is a transverse angular velocity in the direction of the unbalance. (Actually the direction of the unbalance is changing constantly, so that the direction of the angular velocity is a sort of average of the directions which the unbalance had when the two bourrelets cleared the launcher.)

Dynamic unbalance[o] is a slightly more complicated concept and is entirely independent of static unbalance, that is, of the position of the center of mass. In the absence of external forces, the only stable rotational state of a rigid body is rotation about an axis which makes its moment of inertia either a maximum or a minimum. There are in general three such axes at right angles to each other, and they are known as the principal axes of inertia. A perfect spinner would have its principal axis corresponding to minimum moment of inertia coincident with the bourrelet axis, but in general there is a small angle u_D between them, which we shall take as the measure of the dynamic unbalance. If the launcher is tight enough and rigid enough to constrain the round to rotate about its bourrelet axis, the dynamic unbalance creates centrifugal forces causing *opposite* sides of the two bourrelets to press against the launcher, and, when the rocket is freed, the transition from rotation about the bourrelet axis to rotation about the axis of inertia produces a transverse angular velocity, i.e., a mallaunching.

The calculation of the amount of mallaunching with a real launcher is extremely difficult. However, since experiment has shown little difference between dispersions produced by light flexible launchers and heavy rigid ones, it is probably sufficient to make calculations for an absolutely rigid launcher. This is much simpler and is usually what is done. A very elementary calculation will give us an estimate of the mallaunching in this case. We need merely consider the vector s_p representing the spin angular velocity at the moment of launching (in this approximation we assume that the constraint is removed from both bourrelets simultaneously) and resolve it into two components, one parallel to and

[n] Assuming the usual case in which the center of mass is between the bourrelets.

[o] Ordinarily, in the literature, a spinner is said to be dynamically unbalanced when it has either or both of the two types of unbalance.

[redacted]

one perpendicular to the axis of inertia. The latter is the transverse angular velocity.

$$q = s_p \sin u_D \approx s_p u_D.$$

This derivation neglects the details of the interaction between the launcher and the two bourrelets and cannot be expected to give an exact answer, but it does show correctly that the mallaunching due to unbalance is proportional to the spin rate at launching. When the effect of static unbalance is considered also,[12] we find that the mallaunching is given to a good approximation by

$$q = \tfrac{3}{4} s_p u,$$

where

$$u = \sqrt{u_D^2 + 2u_S^2}.$$

The method of combining dynamic and static unbalance by the square root of the sum of the squares assumes that they are randomly oriented relative to each other, and the factors 2 and ¾ come out of more complicated analysis.

We are now in a position to apply the mallaunching formulas to the determination of the dispersion to be expected as a result of dynamic and static unbalance. At the end of Section 25.3.3 we calculated the deflection of our hypothetical HCSR at the end of burning for unit mallaunching from launchers of three different lengths. The mallaunching due to unbalance for a truly zero-length launcher would be zero according to our calculations, but we can apply the zero-length solution approximately to a very short launcher, say 1 ft long, which is just the bourrelet spacing for the 5.0-in./5 HCSR.

For the spin at launching, we have

$$s_p = \frac{2\pi}{\nu}\sqrt{2Gp} = 46\sqrt{p}$$

$$= 46 \text{ radians per second for } p = 1$$

$$= 103 \text{ radians per second for } p = 5$$

$$= 152 \text{ radians per second for } p = 11.$$

If we assume that the total unbalance u is 0.001 radian, the angular velocities of mallaunching produced by the three launcher lengths are respectively 0.0345, 0.077, and 0.114 radians per second. Hence, from the values of θ/q at the end of Section 25.3.3, we calculate the deflections to be

$$4.8 \text{ mils for } p = 1 \text{ ft};$$

$$6.9 \text{ mils for } p = 5 \text{ ft};$$

$$8.1 \text{ mils for } p = 11 \text{ ft}.$$

From these three values it appears that dispersion due to unbalance is small for very short launchers, and that for launchers of practicable length it is a very slowly increasing function of launcher length. These same conclusions are arrived at in a different way in *Rocket Design*,[12] and they appear to be supported by the experimental data. Since very short launchers are not practicable, as we shall see, one more conclusion can be drawn: namely, that if the dispersion of the HCSR is to be much less than 10 mils, considerable care must be taken in balancing it. Thus consider the 6.9-mil figure. To get the lateral dispersion for firing at 45-degree elevation, this is multiplied by $2/\pi$ (because of the random orientation of the unbalance) and by $\sqrt{2}$ (to correct for elevation angle), and the result is 6.2 mils. The actual dispersion of the HCSR is about 50 per cent greater than this, so an unbalance of only 0.001 radian (i.e., 1 mil) will account for most of it. The actual magnitudes of unbalance exhibited by production line spinners are given in Table 4.

TABLE 4. Effect of dynamic unbalance on dispersion of spinners.

Type of rocket	3.5-in./4 GPSR	5.0-in./10 GPSR	5.0-in./14 GASR
Number of rockets	100	93	129
Purely static unbalance (displacement of center of gravity from geometric axis in in.)	0.0035	0.0137	0.0094
Purely dynamic unbalance (angular deviation of dynamic axis from geometric axis in mils)	0.53	0.88	0.97
Mean deflection (mils)	5.9	7.8	6.2
Mean lateral or vertical dispersions (mils)	4.2	5.5	4.4

25.6.2 Effect of Jet Malalignment

The simplest treatment of jet malalignment for spinners is to consider its effect as being equivalent to a mallaunching. This is possible because only for a very short time after it becomes free of the launcher

[redacted]

is the rocket spinning slowly enough for the effect of malalignment to be significant. The relationship between malalignment and equivalent transverse angular velocity of mallaunching, derived in *Exterior Ballistics*,[1b] is approximately

$$q = R_m \frac{\sqrt{5G\nu}}{\pi K^2 \sqrt{2}} E,$$

where R_m is the jet malalignment in feet and E is a dimensionless function of the launcher length and certain constants of the round, which has been calculated and tabulated in *Exterior Ballistics*.[1] For our purpose, we may take E to be given with sufficient accuracy by [p]

$$E = \frac{1}{\sqrt{1 + 20pK^2/\lambda k^2}} \cdot$$

Thus E is equal to unity for a zero-length launcher and decreases rapidly with increasing launcher length, corresponding to the fact that a given malalignment results in less mallaunching on longer launchers, as we would expect, since the rate of spin on leaving the launcher is higher.

For example, in an HCSR of the characteristics given in Table 1, with a malalignment R_m of 0.001 in. (0.0000833 ft), the deflections of the trajectory from launchers of length 0 and 5 ft, will be as follows, in terms of equivalent mallaunching effect:

1. p	ft	0	5
2. R_m	ft	0.0000833	0.0000833
3. $\frac{\sqrt{5G\nu}}{\pi K^2\sqrt{2}}$	ft^{-1}	64	64
4. E	......	1.00	0.238
5. q	radians per second	5.33	1.29
6. θ/q	seconds	0.139	0.089
7. θ	mils	0.74	0.115

In this tabulation, the first five lines are simply data for and evaluation of the preceding equations, to give, in line 5, the transverse angular velocities of mallaunching equivalent, in deflection effect on the trajectory, to the jet malalignment. These quantities are multiplied by the data in line 6, which are the trajectory deflections (angles) per unit angular velocity of mallaunching listed at the end of Section 25.3.3. The figures in line 7 give the trajectory deflection angles, in mils per thousandth of an inch malalignment.

[p] E differs by a constant factor from the function $|E_\infty|$ of *Exterior Ballistics*.[1]

This calculation shows us that for very short launchers, malalignment for spinners would be as serious as for some fin-stabilized rockets (the HVAR deflection, for example, is 1.0 mils per 0.001-in. malalignment). Hence spinner launchers are made just long enough so that the dispersion due to malalignment is considerably smaller than that due to unbalance. Further increasing the launcher length gives little improvement because of the very slow change in unbalance effect with launcher length.

25.6.3 Optimum Spin

It has been mentioned before, but probably cannot be too strongly emphasized, that the attainment of high accuracy, say 5 mils or better, probably requires the use of higher rates of spin than those used in any present ground-fired spinners. Higher spin would reduce the high sensitivity of the present rounds to cross winds. Analysis shows that the wind sensitivity (i.e., the trajectory deviation per unit cross wind) is approximately inversely proportional to the stability factor and hence, other factors being the same, inversely proportional to the square of the spin velocity. When the spin is increased, dispersion due to malalignment decreases. Dispersion due to mallaunching of constant magnitude also decreases, but, as we have seen, the magnitude of the mallaunching is likely to increase. This can probably be mostly compensated by the reduction in launcher length which reduced malalignment effect makes possible. With the wind effect reduced, it would then be profitable to take greater care in balancing the rounds.

On the other hand, such high-stability rounds will not follow a rapidly turning trajectory, so that, in cases where it is required that they do so, one must probably be content with present dispersions.

25.7 SPINNER RANGE CALCULATIONS

For a perfectly launched spinner, the range calculations would differ little from those for a finner. In each case the starting point is the theory of the equivalent shell. The estimation of air drag is easier for a spinner because of the usually simpler shape and lack of lugs and fins, but the drag is a function of yaw angle and, since the spinner does not have an average zero yaw, this relationship

[redacted]

must be known. The difficulty with purely theoretical range calculations is that spinners are almost never perfectly launched, and the direction and magnitude of the mallaunching depend upon the propellant temperature and upon the particular launcher. Thus the only feasible way to construct range tables is to start with experimental data under standard conditions and use the theory to make corrections to other conditions. This is the system adopted in reference 13, and for details of the procedure, the reader is referred to this report. The basic theory is given in *Exterior Ballistics*.[1] Obviously the same remarks apply to the mean deflections as to the ranges.

25.8 TRAJECTORIES OF SPINNERS FIRED FORWARD FROM AIRPLANES

All the characteristic functions for spinner motion which are tabulated in this chapter were calculated on two assumptions: (1) constant stability factor and (2) proportionality of overturning moment to yaw angle. The reason is that the solutions of the equations of motion for the more general case are not possible in terms of functions with which mathematicians are familiar and can be evaluated only by numerical methods.

In ground firing, this rather restricted theoretical treatment covers many cases of interest. Thus, if we stay well below the velocity of sound, the stability factor is very nearly constant during burning and changes very slowly thereafter. Yaws do not exceed about 10 degrees, in which range the nonlinearity of the overturning moment is not great enough to alter the motion significantly. In no case, however, are these conditions true in aircraft firing, so that a rather comprehensive program of computations with a differential analyzer may be required before sighting tables such as those for fin-stabilized aircraft rockets can be made.

When spinners are fired forward from airplanes, they are subjected to large aerodynamic forces as soon as they clear the launcher, while their spin is still small. As a result, their stability factor is below the critical value, and the yaw and transverse angular velocity tend to increase rapidly. Usually the spin increases to the point where the rocket becomes stable again so that the yaw is damped out, but two factors may prevent this. (1) The overturning moment coefficient increases considerably at approximately the velocity of sound; thus reducing the stability factor so that the rocket may not be stable even at the end of burning when the spin is greatest. Thus a spinner having $S = 6$ for ground firing may drop to $S = 2.5$ at the end of burning in forward firing at high airplane speeds. (2) The yaw may build up to the point where the nutations become negatively damped by the Magnus moment. Even though the rocket may recover, it is likely to acquire a rather large deflection during its period of instability, so it is desirable to reduce the duration of this period as much as possible.

From the expression for the stability factor, equation (23) of Chapter 21, it is seen that the most convenient ways to increase the stability of a rocket are to increase its spin or to decrease its transverse radius of gyration, that is, its length. Because the spin is limited by the centrifugal force which the grain can stand, it was found necessary in adapting the 5.0-in. spinner to forward firing to change both the spin and the length. It may also be possible to increase the stability by moving the center of mass of the rocket forward, thus reducing the overturning moment coefficient μ. It must not, however, be moved so far that the Magnus moment reverses, or the rocket will be unstable after burning because of the negative damping of the nutations.

The requirements for a spinner to be fired sideways from an airplane are similar to those for forward firing. If the rocket maintains its orientation during burning, the speed will soon build up to the point where yaw with respect to the air is small even though it may have been nearly 90 degrees at the start. The rocket will then be stable. The condition that its orientation be unchanged is that the transverse angular velocity build up slowly and be small when the rocket has become stable. This requires a large spin velocity and a low overturning moment at large yaw.

When fired backward from aircraft, the rocket is for a time moving base first through the air with decreasing speed so that it becomes stable considerably sooner than when fired forward. No ballistic calculations are possible during this period because the normal airflow from base to nose along the rocket is completely disrupted by the jet. It is difficult to say what is required in this case, since no experimental data are available, but it appears that high spin will be desirable here also.

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25.9 TERMINAL BALLISTICS OF SPIN-STABILIZED ROCKETS[q]

There is a current impression that the underwater and underground trajectories of spin-stabilized rockets and shells are short because of the spin. However a close examination of the question shows that the presence or absence of spin should be of relatively little importance in determining this feature of the terminal ballistics of the projectile. All important differences between the behavior of fin-stabilized rockets and spin-stabilized projectiles are due to other factors, such as the nose shape, the ratio of length to diameter, and the ratio of mass to cross-sectional area. It is probable that the terminal ballistics of spin-stabilized rockets could be improved, if desired, by the use of the proper nose shapes.

The only ways in which spin could modify the terminal ballistics of a projectile are if the nutation and precession associated with the spin modify the motion, or if the spin causes the medium to exert additional forces on the projectile. Now gyroscopic effects are not evident in the usual projectile until it has made three or four revolutions. Over shorter periods it responds to applied forces in essentially the same way as does an unrotated projectile. Hence it seems clear that gyroscopic effects are unimportant during entry. When the projectile is traveling in the bubble under water or in earth, there will be large forces exerted at the nose and tail, but the total torque acting about a transverse axis is extremely small, as is shown by the fact that the angular acceleration about such an axis is small for a fin-stabilized rocket. Hence we should expect no serious gyroscopic effects on the trajectory of a spin-stabilized projectile, provided it has a satisfactory underwater or underground trajectory when not spinning. There might be some tendency for the simple circular trajectory of a nonspinning projectile to be warped into a section of a helix, but the pitch would be long and the total distance traveled the same. It seems safe to assume that the only additional forces and torques of appreciable magnitude exerted by the medium, because of the presence of spin, are a torque tending to decrease the spin about the longitudinal axis.

It follows from these considerations that the theory of underwater ballistics discussed in Chapters 15 and 24 can be applied to spin-stabilized projectiles, as well as to nonspinning projectiles.

Probably the most important factors to be considered in getting satisfactory terminal ballistics in water and earth are the use of such a nose shape and such a ratio of length to diameter that the cross forces and the curvature of the trajectory are small. The next most important factors are the use of a nose shape having a small drag coefficient and the use of a large ratio of mass to cross-sectional area in order to get a small deceleration for a given drag and hence to get a long underwater or underground travel.

Well-designed fin-stabilized rockets tend to be longer and heavier than well-designed spin-stabilized rockets of the same diameter, and spin-stabilized rockets tend to be longer than shells. These characteristics are the result of efforts to secure efficient rocket propulsion, low dispersion, and satisfactory flight. It follows that fin-stabilized rockets will almost always have somewhat longer and straighter underwater and underground trajectories than will spin-stabilized rounds. However the usual spin-stabilized rocket has such a nose shape that its underwater trajectory is much poorer than the optimum set by its length and mass. In the case of the 5.0-in./10 GPSR, the length to diameter ratio is 7, and it is probable that by the use of a suitable nose shape the underwater behavior could be considerably improved. It may prove to be more difficult to get a satisfactory underwater trajectory for the aircraft spinners, since their length to diameter ratio is only 5. The improvement possible will have to be determined by experiment. Shells are usually from 4 to 6 calibers long, and hence it may be difficult to achieve a satisfactory underwater trajectory, particularly since the nose shape is usually chosen to give low drag in air and this tends to give very large cross forces in water. It should be noted, however, that the British seem to have had considerable success in designing shells having a relatively long underwater trajectory.

[q] This section is adapted from an informal memorandum by Leverett Davis, Jr.

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GENERAL BIBLIOGRAPHY OF TECHNICAL REPORTS ISSUED UNDER DIVISION 3, NDRC

Of the many types of technical reports resulting from the work of Division 3 during World War II, the following lists include nearly all those believed to be of continuing general value. As far as possible, the reports are listed under the contracts covering the activities reported. Under "Noncontract Reports" are listed those which cannot properly be credited or limited to any single contract. The scope of the work under each contract is indicated briefly under "Contract Numbers" and at more length in the Introduction to this one-volume Summary Technical Report of Division 3.

In Division 3 (and its predecessor NDRC units, Sections C and H of Division A) the forms of reports and the machinery for their preparation, identification, and distribution varied with time and from one contract to another. Hence, there is some lack of uniformity in the following lists.

Under each contract the OSRD monographs (if any) are listed first, followed by the final reports, then the interim reports, then the periodical reports (if any), and finally miscellaneous other types. The monographs are unclassified publications to be published by the McGraw-Hill Book Company; they cover theory and principles developed during the contract work in fields indicated by their titles. The final reports review and summarize the contract activities and results. On two contracts ultrafinal "summary reports" recapitulate the information presented in many final reports on separate programs.

Most of the interim reports present the status and results of individual research and development projects or programs as of the times when the reports were prepared. These were written as results became available, frequently in response to expressed needs for the information.

Under the larger contracts, periodical reports (weekly, biweekly, or monthly) were prepared, primarily for intracontract and intradivision information and control. These outlined activities and presented test results, usually in substantial detail.

The nomenclature used in titles was that in common use when the reports were written; in some cases it has changed since then.

IDENTIFICATION NUMBERING

All Division 3 reports distributed through or for the Executive Secretary of OSRD bear OSRD numbers included in the lists below. In addition to the OSRD numbers, many of the reports also bear NDRC "A" numbers; all these were edited, reproduced, and distributed by the NDRC (Division A) Technical Reports Section. Most of the reports prepared under Contracts OEMsr-273 and OEMsr-418 had contractors' numbers assigned to them; in some cases these are the only identifications. The "A" series reports are listed with dates of issue, sometimes months after preparation of the manuscript. In most other cases the dates listed are closer to the date of manuscript completion.

AVAILABILITY

Most of the more formal types of Division 3 reports were rather widely distributed to Army and Navy offices concerned with the development of rockets and underwater ordnance. Copies of the monographs will be distributed also to the Army and Navy. Additional copies will be obtainable only by purchase; the publisher has agreed to a 33⅓ per cent discount on purchases by government agencies. With the few exceptions noted in the lists below, several hundred copies of each final report were distributed to the Army and Navy. Fifty to one hundred or more copies of all "A" series reports were so distributed, as were larger numbers of all except the first few of the OEMsr-418 J, LMC, and PMC series. Except where otherwise noted in the lists below, other types of reports were given only limited distribution.

In connection with termination of contracts, all OSRD contractors were required to dispose of all copies of their contract reports, except for record copies. In the case of Contract OEMsr-418, at least one complete set, several nearly complete sets, and large numbers of surplus copies of reports were transferred to the custody of the Naval Ordnance Test Station, Inyokern, California. In the case of Contract OEMsr-273, collections of reports were transferred to the Navy Bureau of Ordnance, for use at the Allegany Ballistics Laboratory (now operated by the Hercules Powder Company) and to the Johns Hopkins University Applied Physics Laboratory. The Bureau of Ordnance took over also the collections of reports associated with the OEMsr-716 work at the University of Minnesota. Other contractors delivered their surplus reports to the Library of Congress, either directly or through OSRD.

Record copies of the more formal Division 3 reports are in the files of Division 3, of the OSRD Executive Secretary, and of the OSRD Liaison Office. Rather complete collections of Division 3 reports are maintained by Office of Naval Research, Navy Department; Research and Development Division, War Department General Staff; Bureau of Ordnance, Navy Department; and Office of the Chief of Ordnance, War Department.

Surplus copies were transferred to the Library of Congress. OSRD reports there are available to meet Army and Navy requests. Photostatic and microfilm copies of unclassified reports are available to the public from the Office of Technical Services, U.S. Department of Commerce.

Of the reports listed below, positive and negative microfilms of those with microfilm numbers are to be deposited with the Coordinator of War Department Libraries, as a supplement to this Summary Technical Report, to meet government needs. Microfilmed reports are listed in the volume bibliography starting on page 347.

SECURITY CLASSIFICATION

Nearly all Division 3 reports were classified when issued. Most of them were confidential. As the result of reclassification review after World War II ended, few of them retain this classification now. Some reports indicating uses and performance of weapons and components involved in continuing Service development programs remain restricted. Most of the reports on standardized or obsolete weapons and on general theory, principles, phenomena, and instrumentation have been declassified; others may be in the future.

RELATED REPORTS

As indicated in the Introduction to this Summary Technical Report volume, work related to that of Division 3 was done by other divisions, the Services, their contractors, and other agencies (including some of the United Kingdom). No adequate attempt can be made in this volume to list or refer to the reports on this work.

NONCONTRACT REPORTS OF DIVISION 3

NDRC No.	*OSRD No.*	
A-4	8	*Jet Acceleration of Armor-Piercing Bombs, as of March 1, 1941*, C. N. Hickman, June 1941.
A-21	24	*The Use of Copper Balls for Measuring Pressures in Combustion Chambers*, C. N. Hickman, Nov. 1941.
A-22	316	*Internal Ballistics of Power Driven Rockets*, E. Lakatos, Dec. 1941.
A-24	320	*The Mechanical Efficiencies of Rockets in Empty Field-Free Space*, J. W. M. DuMond, Jan. 1942.
A-25	323	*Microtome Sections of Ballistite Powder*, A. J. Dempster, Dec. 1941.
A-27	311	*Rocket Targets as of November 1, 1941*, A. J. Dempster, Dec. 1941.
A-28	345	*The Trajectories of Target Rockets*, A. J. Dempster, Jan. 1942.
A-36	466	*A Vacuum Press for the Extrusion of Solventless Double-Base Ballistite for Use in Rockets*, J. W. M. DuMond, Mar. 1942.
A-69	691	*Jet Acceleration Tests of the 14-in. Armor-Piercing Bomb*, C. N. Hickman, July 1942.
A-70	673	*A 4½-in. High-Explosive Rocket Shell for Projection from Airplanes*, C. N. Hickman and L. A. Skinner, July 1942.
A-78	769	*Inspection and Testing of a ⅞-in. Stick Powder*, J. Beek, Jr., Aug. 1942.
A-102	943	*The Rate of Burning of Double-Base Powders and the Possible Effects of Change in Nitroglycerin and Total-Volatiles Content on the Burning of Jet Propulsion Tube Power*, R. E. Gibson, Oct. 1942.
A-117	1070	*Microscopic Structure and the Development of Flaws in Extruded Grains of NT Smokeless Powder*, C. P. Saylor, Nov. 1942.
A-127	1136	*Microscopic Examination of Extruded Smokeless Powders*, C. P. Saylor, Dec. 1942.
A-150	1264	*The Design of Granulation for Rocket Powder*, J. Beek, Jr., Feb. 1943.
A-166	1359	*Some Problems of Heat Transfer in Rockets*, J. Beek, Jr., Apr. 1943.
A-178	1473	*Studies of the 4.2-in. Chemical Mortar*, A. R. T. Denues, Apr. 1943.

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NDRC No.	OSRD No.	
A-185	1466	*Abstracts of Technical Reports on Rockets, Vol. 1: NDRC Publications, as of May 24, 1943*, May 1943.
A-197	1590	*Heats of Combustion and Formation of Diethylphthalate, Dibutylphthalate, Dinitrotoluene, Diethyldiphenylurea and Nitroguanidine*, E. J. Prosen and R. Gilmont, July 1943.
A-198	1649	*Abstracts of Technical Reports on Rockets, Vol. 2: Air Corps Jet Propulsion Research Reports*, July 1943.
A-202		*Project Summaries for Division 3, Special Projectiles, as of March 1943.*
A-215	1801	*Interior Ballistics of Recoilless Guns*, J. O. Hirschfelder, R. B. Kershner, C. F. Curtiss, and R. E. Johnson, Sept. 1943.
A-267	3512	*A Microscopical Study of German Powder from a 21-cm Aircraft Rocket*, C. P. Saylor, Apr. 1944.
A-285	3932	*Heats of Combustion of Celluloses and Nitrocelluloses*, R. S. Jessup and E. J. Prosen, July 1944.
A-287	4033	*A Study of the Effect of Radiation on the Burning of Rocket Powder*, J. Beek, Jr.

NONCONTRACT MEMORANDA OF DIVISION 3

A-1L		*Derivations of Formulas Used in Computing Effective Gas Velocity and Rocket Velocity from Measured Impulse*, J. W. M. DuMond, Nov. 1941.
A-4M—A-17M	28	*Notes and Tests of the Design and Performance of Jet Propelled Devices, November 8, 1940 to July 18, 1941*, C. N. Hickman, Sept. 1941.
A-18M—A-27M	29	*Notes on the Design and Performance of Jet Propelled Devices to November 1, 1941*, C. N. Hickman, Nov. 1941.
A-28M—A-30M	30	*New Equipment for Measuring and Recording the Pressure of Powder Gases in Rocket Chambers and the Thrusts Exerted by Blocked Rockets, Each as a Function of Time*, J. W. M. DuMond, Nov. 1941.
A-38M—A-39M	433	*New Gages for Measuring the Thrusts of Rockets*, C. N. Hickman, Feb. 1942.
A-43M	665	*A Partial Burning Powder Tester*, C. N. Hickman, June 1942.
A-46M	701	*Design of Rocket Targets Adopted by the Army Ordnance Department*, L. A. Skinner, July 1942.
A-55M	924	*Static Tests of the 12-in. Jet-Accelerated Armor-Piercing Bomb*, C. N. Hickman, Oct. 1942.
A-57M	965	*Remarks on the Applications and Performance of Rockets*, C. N. Hickman, Oct. 1942.
A-58M	1022	*Thermal Ignition and Arming Elements for Use with Rockets*, C. N. Hickman, Nov. 1942.
A-107M	4830C	*Annotated Bibliography of NDRC Technical Reports and Memorandums of Division 3, Including a Listing of Pertinent Reports Issued by Contractors of Division 3, as of September 15, 1944.*

CONTRACT REPORTS FROM DIVISION 3

Contract No.	
OEMsr-250	California Institute of Technology Included in OEMsr-418. See below.
OEMsr-256	Western Electric Company (Bell Telephone Laboratories) Many reports, listed below.
OEMsr-273	The George Washington University Many reports, listed below.
OEMsr-416	Hercules Powder Company Final Report, July 1, 1943. (Limited distribution to Services.) *Development of Jet Propulsion Powders*, C. W. Gault.
OEMsr-418	California Institute of Technology Many reports, listed below.
OEMsr-671	Budd Induction Heating, Inc. Final Report, Sept. 1943. (Limited distribution to Services.) *Investigation M-7 Report on Development of 4½-in. Rocket under Office of Emergency Management and Scientific Research.* Supplement to this—*Operation Drawings.*
OEMsr-702	California Institute of Technology Final Report, Jan. 1943. (Limited distribution to Services.) *Investigations of Double-Base Powders*, Linus Pauling. (See also later Division 8 reports on Contract OEMsr-881, under which similar work was continued at CIT.) Interim Reports. (Written under the contract, edited and issued by NDRC.)

NDRC No.	OSRD No.	
A-124	1103	*Investigations of Double-Base Powders: Spectrophotometric Studies, I*, R. B. Corey, A. O. Dekker, and A. M. Soldate, Dec. 1942.
A-128	1151	*X-Ray Diffraction Studies of Molecular Orientation in Double-Base Smokeless Powders Made by the Solvent and Solventless Processes*, H. A. Levy, Dec. 1942.

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A-132 1152 *Chromatographic Studies of Double-Base Powder I*, R. B. Corey, R. Escue, A. L. LeRosen, and W. A. Schroeder, Jan. 1943.

A-151 1265 *Measurements of pH on Double-Base Powders*, R. B. Corey, C. Green, and H. Levy, Feb. 1943.

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Contract No.

OEMsr-716 University of Minnesota

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Final Report Supplement. *Visual Studies of Propellant Burning.* (A two-reel Kodachrome film given limited distribution.)

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A-130 1188 *The Available Literature on the Mechanism of Combustion of Double-Base Powders*, B. L. Crawford, Jr., and C. Huggett, Jan. 1943.

A-171 1370 *The Ignition by Radiation and Fissuring of Double-Base Powder*, B. L. Crawford, Jr., C. Huggett, H. S. Isbin, and J. J. McBrady, Apr. 1943.

A-200 1713 *Determination of Ignition Temperatures of Double-Base Powders*, B. L. Crawford, Jr., and H. S. Isbin, July 1943.

A-268 3544 *Observations on the Burning of Double-Base Powders*, B. L. Crawford, Jr., C. Huggett, and J. J. McBrady, Apr. 1944.

A-286 4009 *Direct Measurement of Burning Rates by an Electric Timing Method*, B. L. Crawford, Jr., and C. Huggett, Aug. 1944.

Contract No.

OEMsr-733 Duke University

Final Report, November 1944. *Determination of the Linear Burning Rates of Propellants from Pressure Measurements in the Closed Chamber*, L. G. Bonner.

Interim Report, June 1944. Title and author as above.

OEMsr-762 University of Wisconsin

Final Report, 1945. (Edited and issued by NDRC.)

NDRC No. *OSRD No.*

A-485 6559 *Studies of the Mechanism of Burning of Double-Base Rocket Propellants*, Farrington Daniels and associates.

Interim Reports. (Prepared under the contract, edited and issued by NDRC.)

A-173 1362 *Testing Powder Grains for Fissures with Special Emphasis on Nonvisual Methods*, Farrington Daniels and R. E. Wilfong, Apr. 1943.

A-243 3206 *The Mechanism of Powder Burning*, University of Wisconsin, Jan. 1944.

Contract No.

OEMsr-947 Catalyst Research Corporation

Final Report, Sept. 28, 1943. (Limited distribution to Services.) *A Delay Element for Use with Rocket Motors*, John P. Woolley.

OEMsr-968 Budd Wheel Company

Final Reports, late 1945.

OSRD No.

6132 *Rockets for Cast Propellants.*

6133 *T-59 High Velocity Rocket Grenade (Super Bazooka).*

6134 *Rocket Launchers.*

6135 *Flare Rockets.*

6136 *Airborne Flame Thrower.*

6137 *Tank Borne Flame Thrower.*

6138 *Portable Powder Pressurized Flame Thrower.*

6139 *115-mm Rocket (OD-161, NO-245, Army Int. only).*

6140 *High-Performance Rocket—Design Studies.*

6141 *Multiple Powder Charge Launcher for JB-2.*

6142 *Lightweight 4.2-in. Recoilless Chemical Mortar.*

6143 (This number canceled. The report, on the driver rocket, is included in OSRD 6142.)

6144 *Extension of Range of the 4.2 inch Chemical Mortar "M–."*

6145 *Development of a New 4.2 inch Chemical Mortar of Radical Design.*

6146 *Step Motor Rockets.*

6147 *Rocket Powder Traps.*

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OSRD No.

6148 *4½ inch Rotated Rockets.*
6149 *60-mm Recoilless Mortar.*
6150 *81-mm Recoilless Mortar.*
6151 *12" Jet Propelled AP Bomb.*
6152 *Miscellaneous Development of Rockets and Accessories.*
6153 *Summary Report of Rocket Developments.*

Contract No.
OEMsr-256 Western Electric Company (Bell Telephone Laboratories)
Final Reports.

OSRD No.

6154 *Engineering and Material Service*, S. R. Avella.
6155 *Mechanical Arming Propeller for 12" Jet-Accelerated AP Bomb*, R. F. Mallina.
6156 *Launchers and Improved Components for 4.5-in. Rocket*, J. M. Dietz, R. F. Mallina, C. F. Spahn, and J. M. Melick.
6157 *Development of Ribbon Frame Camera*, F. L. McNair and F. Reck.
6158 *Ripple Firing Mechanisms for Launching Rockets*, D. D. Miller and T. H. Guettich.
6159 *Powder Trapping and the Extrusion of Rocket Powders*, R. Burns.
6160 *The Development, Annealing and Calibrating of Copper Tarage Balls*, J. R. Townsend.
6161 *The Development of Apparatus for Recording Pressure vs Integral of Pressure*, K. S. Dunlap.
6162 *Rocket Launchers for Use on Aircraft*, J. M. Dietz, C. A. Hasslacher, and J. H. Mogler.
6163 *Design and Production of Amplifier Calibrators*, J. S. Garvin.
6164 *The Design of a 4.2-in. Recoilless Mortar Mount*, J. M. Dietz.
6165 *Airborne Flame Thrower Firing Circuits*, P. E. Buch.
6166 *The Firing of Rockets by Induction Methods*, J. M. Melick.
6167 *Tank Borne Flame Thrower Firing Circuits*, P. E. Buch.
6168 (This number canceled. The report, on an electromagnetic fuze, is included in W-6.1, one of the final reports listed below under George Washington University Contract OEMsr-273.)
6169 *X-Ray Diffraction Photograph Investigations of Sheet Powder*, W. O. Baker and N. R. Pape.
6170 *Summary Report on Rocket Developments*, S. R. Avella.
Interim Reports. (Written under the contract, edited and issued by NDRC.)

NDRC No.	*OSRD No.*	
A-134	1212	*Basic Flow Properties of Powders of Various Compositions*, R. Burns, Jan. 1943.
A-187	1484	*Firing Mechanism for the 7-in. Chemical Rocket Projector*, R. F. Mallina and P. T. Higgins, May 1943.
A-196	1605	*The Ribbon-Frame Camera*, F. Reck, July 1943.
A-79M	3035	*Mechanical Fuze for 12-in. Armor-Piercing Bombs*, R. F. Mallina, Dec. 1943.
A-80M	3082	*Vertical Rocket Launcher for Airplanes*, R. F. Mallina, Jan. 1944.
A-81M	3083	*Jungle Launchers*, R. F. Mallina, Jan. 1944.
A-89M	3446	*Light-Weight Rocket Projectors*, R. F. Mallina and J. M. Dietz, April 1944.

Contract No.
OEMsr-273 The George Washington University (includes Allegany Ballistics Laboratory [ABL]).
Monograph (unclassified). *Mathematical Theory of Rocket Flight*, J. B. Rosser and R. R. Newton, to be published by the McGraw-Hill Book Company, Inc.
Final Reports, 1946.

GWU No. [a] *OSRD No.*

OSRD numbers in the block 5771-5897 which are absent from the list below were canceled.

GWU No.	*OSRD No.*	
B-1.2	5872	*Drag of the Propellant Gases on the Powder Charge in Rockets, Part I. Simple Theory*, F. T. McClure and J. B. Rosser; *Part II. Experimental Measurements*, J. F. Kincaid and F. T. McClure.
B-2.1	5861, 5877	*Theory and Application of* $\int_0^z e^{-x^2}\,dx$ *and* $\int_0^z e^{-p^2y^2}dy \int_0^y e^{-x^2}dx$, *Part I. Methods of Computation*, J. B. Rosser; *Part II. Applications to the Exterior Ballistic Theory of Rockets*, G. L. Gross, J. B. Rosser, and E. M. Cook.

[a] The GWU numbers group the final reports by type of subject matter, as follows:
B indicates ballistic theory and design.
W indicates weapon development.
P indicates propellant development.
J indicates instrumentation.

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GWU No.	*OSRD No.*	
B-2.2	5878[b]	(This is the monograph listed above.)
B-2.3	5879[c]	*Motion of a Spin-Stabilized Rocket during the Burning Period*, W. J. Harrington.
B-2.4	5888	*Flight Ballistics Involved in the Use of Rocket-Towed Devices*, W. J. Harrington.
B-2.5	5882	*Correlation of Wind Tunnel Data on Rockets*, N. G. Gunderson and Seymour Sherman.
B-2.6	5883	*The Relation of Manufacturing Tolerances to Dispersion of Fin-Stabilized Rockets*, R. R. Newton and M. Goldman.
B-2.7	5884	*Exterior Ballistics of the Cable Bomb*, G. L. Gross and J. B. Rosser.
B-3	5886	*Some Problems of Heat Transfer in Rockets*, J. Beck, Jr., J. B. Rosser, and Harry Siller.
B-3.1	5887[c]	*Temperature Transients in Walls of Rocket Chambers*, E. A. Cook and E. H. deButts, Jr.
B-3.2	5889	(Canceled.)
B-4	5890	*Propellant Charge Design of Solid Fuel Rockets*, W. H. Avery and J. Beek, Jr.
B-5	5891	*The Design of Metal Components for Rocket Motors*, H. C. Stumpf and G. W. Engstrom.
B-5.1	5893	*Investigation of Fiberglas Laminates as Materials for Rocket Motors*, J. Beek, Jr., and J. F. Kincaid.
B-6	5897	*Theoretical Studies of Long-Range and High-Altitude Rockets*, J. B. Rosser, F. T. McClure, C. N. Hickman, and Nancy Marmer Thompson.
W-1	5769	*The Jet-Accelerated Armor-Piercing Bomb*, C. N. Hickman.
W-2	5775	*Rocket Targets Developed for the Army Ordnance Department*, C. N. Hickman.
W-3	5771	*High-Explosive Anti-Tank 2.36-in. Rocket (Bazooka)*, C. N. Hickman and S. Golden.
W-3.1	5773	*Interim Ballistic Studies*, S. Golden.
W-3.3	5776	*The Development of the T-12 Grenade*, D. M. Brasted.
W-3.4	5589	*Development of T4 Powder Charge for M6A3 Rocket Grenade* (issued by Division 8), R. Lumry and L. Streff.
W-4	5777	*Improvement of Components for 4.5-in. Rocket, M8*, D. W. Osborne and B. Weissmann.
W-5	5778[d]	*Rocket Flares*, A. Kossiakoff, N. T. Grisamore, and J. Beek, Jr.
W-6	5779	*T-59 High Velocity Rocket Grenade, Part I. Characteristics and Performance*, S. Golden; *Part II. Internal Ballistics of Laminated Charges*, S. Golden; *Part III. Ignition*, W. P. Spaulding and L. E. Morey; *Part IV. Propellant Loss*, S. Golden, L. E. Morey, and W. P. Spaulding.
W-6.1	5881	*A Point-Initiating Base-Detonating Electromagnetic Fuze*, Allegany Ballistics Laboratory, Bell Telephone Laboratories, and Explosives Research Laboratory.
W-6.2	5780	*The Follow-Through Rocket Grenade, T1*, W. P. Spaulding and S. Golden.
W-7	5794	*Step Motor Rockets*, C. N. Hickman and J. M. Woods.
W-8	5781	*115-mm Aircraft Rocket*, R. E. Gibson and A. Kossiakoff.
W-8.1	5784	*Development of Propellant Charge for 115-mm Aircraft Rocket*, J. Beek, Jr., R. L. Arnett, G. W. Engstrom, M. Goldman, and A. Kossiakoff.
W-8.2	5785	*Development of Rocket Motor for 115-mm Aircraft Rocket*, A. Kossiakoff and G. W. Engstrom.
W-8.3	5786	*Development of Heads and Fuzes for 115-mm Aircraft Rocket*, M. J. Walker, F. T. McClure, and A. Kossiakoff.
W-8.4	5788	*Development of a High-Performance Composite-Propellant Charge for 115-mm Aircraft Rocket*, R. Lumry and L. Streff.
W-9	5789	*Extension of Range of the 4.2-in. Chemical Mortar M2*, G. C. Bowen, C. F. Curtiss, A. R. T. Denues, and R. B. Kershner.
W-9.1	5792	*Summary of Interim Ballistic Studies of the 4.2-in. Chemical Mortar*, A. R. T. Denues.
W-9.2	5694	*Tests of Various Methods of Obtaining Rotation of the 4.2-in. Chemical Mortar Shell*, G. C. Bowen.
W-9.3	5811[d]	*Driver Rocket for the 4.2-in. Chemical Mortar*, G. C. Bowen and R. B. Kershner.
W-9.4	5790	*Development of a New 4.2-in. Chemical Mortar of Radical Design*, T. R. Paulson.
W-10	5791	*Recoilless 4.2-in. Chemical Mortars, Part I. The Development of 4.2-in. Recoilless Chemical Mortar, E34R1*, R. B. Kershner; *Part II. Corrective Development of 4.2-in. RCM Taken from Final Summary Report, No. 12, ABL-CWS 4.2 Mortar Group*, A. R. T. Denues; *Part III. Development of a Cartridge Ignition System*, J. M. Woods; *Part IV. Driver Rocket Development*, S. Golden, J. Levin, and F. Culp.
W-11	5800	*Propellant Charge Development for 4.5-in. Spinner Rockets, T38E5, T105, T110*, D. M. Brasted and S. D. Brandwein.
W-13.1	5795	*Rocket for the Anti-Personnel Mine Clearing Snake, M1*, C. A. Boyd and R. J. Bond.
W-13.2	5796	*Investigations of the Use of Rockets to Dispense Mine Clearing Hose*, S. D. Brandwein, C. A. Boyd, and W. J. Harrington.

[b]OSRD number 5878 was assigned to this report (B-2.2), then canceled when its publication as an OSRD monograph was approved.
[c]Not issued, as of January 1947. Manuscripts loaned by OSRD to Navy Bureau of Ordnance for completion of editing, printing, and distribution by Johns Hopkins University Applied Physics Laboratory.
[d]Manuscripts not completed, but may be issued under above auspices.

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GWU No.	*OSRD No.*	
W-13.3	5799	*The Rocket for the Projected Line Charge*, C. A. Boyd, W. J. Harrington, and D. Leenov.
W-13.4	5798	*Rocket for Projecting Detonating Cable*, C. A. Boyd, D. Leenov, and W. J. Harrington.
W-13.5	5801	*The Rocket for Towing Bangalore Torpedoes*, R. J. Bond and C. A. Boyd.
W-14	5802	*60-mm Recoilless Mortar*, S. Golden and N. T. Grisamore.
W-15	5803[d]	*81-mm Recoilless Mortar*, R. B. Kershner and E. J. Moore.
W-16.1	5804	(Canceled.)
W-16.2	5805	*Portable One-Shot Flame Thrower*, R. E. Hunt.
W-16.3	5880	*Development of Portable Smokeless Powder-Operated Gas Generator for Pressurizing M2-A2 Flame Throwers*, A. S. Collins and A. A. Nellis.
W-16.3s	5806	*Production Model of Portable Smokeless Powder-Operated Gas Generator for Pressurizing M2-A2 Flame Throwers* (Supplementary Report), Robert Lee James.
W-16.4	5807	(Canceled.)
W-17	5808	*A Gas Generator for a Small Turbine*, S. S. Penner and A. J. Madden.
W-18.1	5812	*Booster Launcher for Testing of Aircraft Rockets*, M. J. Walker.
W-18.2	5813	*Spiral Launching of 4.5-in. Rockets*, R. R. Newton.
W-18.3	5814	*Induction Firing for Rockets*, C. F. Bjork and M. Bondy.
W-18.4	5787	*The Development of Rocket Fins and Lug-Band Kits for Use with the Flush-Mount Launcher on Aircraft*, G. W. Engstrom and R. L. Beddoe.
W-19	5815	*The Jet-Assisted Take-Off Unit*, L. G. Bonner and W. H. Avery.
W-20	5818	*A Multiple-Cartridge Launcher for the JB-2*, R. B. Kershner, C. F. Curtiss, V. D. Russillo, and C. N. Hickman.
W-21	5793	*Design of the High-Velocity Rocket (VICAR)*, J. Beek, Jr., R. J. Thompson, and R. R. Newton.
W-21.1	5820	*Small-Caliber High-Velocity Rocket (CURATE)*, R. J. Thompson, D. Brewer, and R. R. Newton.
W-22	5821	*The Bumblebee Rocket Motor*, S. S. Penner.
W-23	5822	*Rocket-Projected Special-Purpose Bombs, Part I. Cable Bomb*, A. Africano and J. B. Rosser; *Part II. Rocket Projection of Incendiary Evaluation Bomb*, S. Shulman; *Part III. Short-Range Rocket-Projected Demolition Bomb*, J. F. Kincaid.
W-24	5867	*Powder-Driven Post-Hole Digger (Donnerkiel)*, R. B. Kershner.
P-1	5827	*Burning Rate Studies of Double-Base Powder*, W. H. Avery; *Part I. Alternate Solventless T2 Powders*, R. E. Hunt and M. N. Donin; *Part II. Slow-Burning Powders*, R. E. Hunt and M. N. Donin; *Part III. Temperature Coefficients of Standard Propellants and Promising Experimental Powders*, R. E. Hunt and M. N. Donin; *Part IV. Compression Molded Double-Base Powders*, M. N. Donin.
P-1.1	5831	*Erosive Burning of Double-Base Powders*, R. J. Thompson and F. T. McClure.
P-1.2	5833	*Determination of Burning Rates of Certain Powders by the Strand Technique*, J. J. Donovan.
P-1.3	5816	*Determination of Burning Rates from Pressure-Time Relations in Closed Chambers*, L. G. Bonner.
P-1.4	5824	*Effect of Pressure and Temperature on the Rate of Burning of Double-Base Powders of Different Compositions*, W. H. Avery, R. E. Hunt, and L. D. Sachs.
P-2	5817	*Studies of Radiation Phenomena in Rockets, Part I. A Study of the Effect of Radiation on the Burning of Rocket Powder*, J. Beek, Jr.; *Part II. Influence of Radiation upon the Burning of Rocket Propellants*, W. H. Avery; *Part III. A Theory of the Effect of Radiation on the Constant Pressure Burning Rate of Powders*, M. J. Dresher and F. T. McClure; *Part IV. Radiation Phenomena in Rockets*, S. S. Penner.
P-2.1	5832	*Flame Temperature and Radiation Studies in Rockets*, R. S. Craig.
P-3	5828	*The Reduced Specific Impulse of Ideal Gases*, N. Marmer and F. T. McClure.
P-3.1	5829	*A Comparison of the Specific Impulse of Four Double-Base Rocket Propellants*, J. P. Rappolt and J. Beek, Jr.
P-3.2	5830	*Impulse Determinations of Rockets by Means of Rotating Systems*, S. S. Penner.
P-4	5834	*Restriction of Powder Burning*, A. Turk, L. G. Bonner, A. J. Madden, J. J. Donovan, and W. H. Avery.
P-5	5837	*A Study of Ignition in the 2.36-in. Rocket Grenade*, R. S. Craig and L. D. Sachs.
P-6	5841	*Determination of Energies of Explosion of Propellant Powders*, J. J. Donovan.
P-6.1	5842	*Certain Special Methods for the Chemical Analyses of Double-Base Powder*, J. J. Donovan.
P-7	5844	*Dry Extrusion of Powder at Allegany Ballistics Laboratory*, H. Higbie and G. F. Padgett.
P-8	5845[c]	*Physical Properties of Propellants*, H. Higbie.
P-9	5851	*Formulation of Manufacturing Specifications for Solid Propellants*, R. L. Arnett.
P-10	5852	*Miscellaneous Propellant Studies, Part I. Investigation of Some Special Propellant Charge Designs*, L. G. Bonner; *Part II. Utilization of Magnesium as a Rocket Fuel*, S. Golden and W. P. Spaulding.

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GWU No.	OSRD No.	
P-10.1	5624	*Ballistic Characteristics and Rocket Design Data for Extruded Composite Propellants*, R. Lumry and L. Streff.
P-10.2	5853	*Captured Enemy Propellants*, M. N. Donin and J. J. Donovan.
J-1	5855	*Static Range Operational and Fire Control Equipment for Rocket Research*, N. E. Alexander and C. M. Lathrop.
J-1.1	5856	*The Manufacture of Wire Strain Gages for the Measurement of Pressure as Applied to Rocket Research*, N. E. Alexander.
J-1.2	5843	*The Manufacture of Wire Strain Gages for the Measurement of Thrust as Applied to Rocket Research*, N. E. Alexander.
J-1.3	5857	*A-C Bridge and Preamplifier for Strain-Gage Measurement of Pressure and Thrust*, N. E. Alexander.
J-1.4	5846	*Standard Automatic Calibrator for A-C Bridge and Amplifiers*, N. E. Alexander.
J-1.5	5858	*Two-Channel Ballistics Camera*, N. E. Alexander.
J-1.6	5869	*Standard Frequency Oscillators, Tuning-Fork Type*, N. E. Alexander.
J-1.7	5870	*Audible Null Indicator for 15,000-cycle/sec A-C Bridge Equipment*, N. E. Alexander.
J-1.8	5862	*Calibration Equipment for Pressure and Thrust Wire Strain Gages*, N. E. Alexander and C. M. Lathrop.
J-2	5859	*Apparatus for the Recording of Pressure versus Pdt*, S. Golden and C. M. Lathrop.
J-2.1	5847	*Electronic Blastmeter*, S. Shulman and W. H. Barber.
J-2.2	5865	*Miscellaneous Experimental Electronic Pressure Recorders for Rocket Research, Part I. Electronic Sweep Pressure-Time Recorder; Part II. Electronic Maximum Pressure Indicator; and Part III. Wide Range Pressures versus Pdt Recorder*, N. E. Alexander and W. H. Barber.
J-3.1	5860	*Bourdon System for Pressure Measurement*, R. E. Hunt and W. H. Avery.
J-3.2	5864	*Piston Thrust Gage*, A. Africano.
J-3.3	5863	*Copper Ball Crusher Gages*, A. Africano.
J-4	5868	*High-Speed Temperature-Time Recorders Employing an Electronic Inverter*, N. E. Alexander.
J-5	5849	*X-Ray Photography of Burning Rocket Propellants*, S. Golden, R. S. Craig and W. P. Spaulding.
J-6	5871	*Application of the Optical-Lever Principle to Following Projection Tip-Off and Yaw Motions During Early Burning of Rockets*, M. R. Goff.

Periodical Reports

Weekly Progress Reports, ABL-WPR 1 (March 30, 1944) through ABL-WPR 63 (Aug. 15, 1945).

Progress Reports of the Jet Propulsion Research Laboratory, Naval Powder Factory, Indian Head, Maryland, 1941 to 1944.

Supplements to ABL *Weekly Progress Reports*

Suppl. No.	OSRD No.	
1	3880	*Influence of Radiation upon the Burning Rates of Propellants*, W. H. Avery, July 8, 1944.
2	3960	*Preliminary Report on the Motion of a Fin-Stabilized Rocket During the Burning Period*, R. R. Newton and J. B. Rosser, Aug. 5, 1944.
3	4074	*Preliminary Report on the Motion of a Fin-Stabilized Rocket During the Burning Period, Part II*, R. R. Newton and J. B. Rosser, Aug. 26, 1944.
4	4466	*Status Report on the T-59 High-Velocity Rocket Grenade*, S. Golden, L. E. Morey, and W. P. Spaulding, Nov. 25, 1944.
5	4487	*Investigations of a Proposed Liquid Carbon Dioxide Rocket*, A. Africano and F. T. McClure, Dec. 2, 1944.
6	4568	*Revisions and Corrections to NDRC Formal Report No. A225*, W. H. Avery, R. E. Hunt, and L. D. Sachs, Dec. 23, 1944.
7	4569	*The Rocket Motor for Mine Clearing Snake, M-1; Development and Current Status*, C. A. Boyd and R. H. Bond, Dec. 23, 1944.
8	4694	*A Method of Induction Firing for Rockets*, C. F. Bjork and M. Bondy, Jan. 20, 1945.
9	4942	*Effect of Pressure and Temperature on the Rate of Burning of Double-Base Powders of Different Compositions: II*, W. H. Avery, R. E. Hunt, and L. D. Sachs, Mar. 2, 1945.
10	4905	*Tables of Functions Related to the Fresnel Integrals*, J. B. Rosser, E. M. Cook, and G. L. Gross, Mar. 10, 1945.
11	4963	*Diffusion of Nitroglycerin in Wrapped Powder Grains*, S. Penner and S. Sherman, June 5, 1945.
12	4964	*Apparatus for the Recording of Pressure vs Pdt*, S. Golden and C. Lathrop, June 5, 1945.
13	5251	*Radiation Phenomena in Rockets*, S. S. Penner, Oct. 16, 1945.
14	3932	*The Status of ABL Projects as of VJ-Day*, ABL Staff, Sept. 3, 1945.

Interim Reports. (Written, in most cases, under the contract, edited and issued by NDRC.)

A-81	793	*Tests of the 4.2-in. Chemical Mortar*, A. Africano and S. Golden, Aug. 1942.
A-93	888	*The Fissuring of Translucent Double-Base Powders at Low Pressures*, A. Africano, Sept. 1942.
A-97	929	*Pressure Relationships Holding Within Small Rockets*, S. Golden, Sept. 1942.

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Suppl. No.	OSRD No.	
A-98	920	*Tests of Cemented Ball Powder Charges*, M. Walker and A. Africano, Sept. 1942.
A-107	1014	*Thermodynamic Properties of Special Double-Base Powders*, D. W. Osborne, F. T. McClure, and J. O. Hirschfelder, Oct. 1942. (See also Division 1 Report A-116, OSRD 1087, *Thermodynamic Properties of Propellant Gases*, J. O. Hirschfelder, F. T. McClure, C. F. Curtiss, and D. W. Osborne, Nov. 1942. Both came mainly from Division 1 work at the Geophysical Laboratory of the Carnegie Institution of Washington under Contract OEMsr-51.)
A-113	992	*Jet Propelled Illuminating Flare*, W. E. Jeremiah, Nov. 1942.
A-119	1079	*Rockets for Assisted Take-Off of Airplanes; Their Use and the Prevention of Blast Injury to the Airplane*, Leo Maas, Jr., Nov. 1942.
A-123	1100	*Dry Extrusion of Double-Base Powder at Indian Head*, H. E. Higbie, Dec. 1942.
A-133	1226	*Dry Extrusion of Double-Base Powder at Indian Head, II. Extrusion of Solventless Sheet Powder of the Russian Formulation*, H. Higbie, Jan. 1943.
A-136	1142	*Burning Characteristics of Russian Powders*, C. F. Bjork and A. Africano, Jan. 1943.
A-153	1303	*The Estimation of Pressure-Time Relations Obtaining in Powder Driven Rockets*, S. Golden, Mar. 1943.
A-184	1460	*Dispersion of a Rotating Rocket*, C. H. Dowker, May 1943.
A-203	1658	*Extrusions of Double-Base Powder at Indian Head*, G. J. Padgett, July 1943.
A-205	1703	*Tests of Rotating Budd 4½ in. Rockets*, D. W. Osborne and M. J. Walker, Aug. 1943.
A-220	1886	*The Extrusion of Dried Solvent Processed Double-Base Powder at Indian Head*, H. Higbie, Sept. 1943.
A-225	1993	*Effect of Pressure and Temperature on the Rate of Burning of Double-Base Powders of Different Compositions*, W. H. Avery and R. E. Hunt, Oct. 1943.
A-231	2085	*Relations Between a Rocket and Its Equivalent Shell*, J. B. Rosser, Nov. 1943.
A-247	3232	*Fin Opening*, J. B. Rosser, Feb. 1944.
A-251	3280	*Instructions for Use of the Ribbon-Frame Camera*, M. J. Walker, Feb. 1944.
A-65M	1156	*Some Effects of Composition, Powder Temperature and Radiation on the Rate of Burning of Double-Base Powders*, W. H. Avery, Jan. 1943.
A-72M	1623	*Changes in the Spider Design in the Extrusion Dies Used at Indian Head*, H. E. Higbie, July 1943.
A-75M	2069	*A Less Regressive Design for Powder Grains*, B. Kelly, F. T. McClure, and J. B. Rosser, Nov. 1943.
A-76M	2069	*Theoretical Study of the Validity of a Certain Method of Determining a Burning Law*, B. Kelly, R. B. Kershner, F. T. McClure, and J. B. Rosser, Nov. 1943.
A-88M	3429	*Assembly Operations for Bayonet Igniter Model No. 2*, J. W. Burns, Mar. 1944.

ABL SPECIAL REPORTS

GWU No.	OSRD No.	
ABLsr-1	3711	*Rocket Fundamentals*, May 1944. (Superseded by ABLsr-4.)
2		*Technical Conference on Plane to Ground Rocket Tests* (Preliminary), Apr. 1944.
2 (Revised)		*Technical Conference on Plane to Ground Rocket Tests*, June 1944.
3		*Tables of Ranges and Times of Flight for Certain Values of Angle of Departure, Muzzle Velocity, and Ballistic Coefficient*, G. L. Gross and W. P. Spaulding, June 1944.
3 (Revised)	5440	*Tables of Ranges and Times of Flight for Projectiles with Small Ballistic Coefficients*, Aug. 1945.
4	3992	*Rocket Fundamentals*, Dec. 1944.
5		*Description and Facilities of the Allegany Ballistics Laboratory*, Cumberland, Md.
6	5231	*Description and Instructions for the Use of Jet-Assisted Take-Off Unit Model 8AE-1000-H5*, June 1945.
8	5394	*Preliminary Description of the E37 4.2-in. Chemical Mortar*, Aug. 1945.
10	5548	*The Reduced Specific Impulse of Ideal Gases*, Nancy Marmer and F. T. McClure, Oct. 1945.
11	5879	*Demonstrations and Descriptions of New 4.2-in. Chemical Mortars of Radical Design*, prepared by the CWS 4.2 Chemical Mortar Group at ABL, Nov. 1945.
13	6299	*Program of Events for Press Day at ABL, Nov. 8, 1945, and Memorandum Describing Laboratory Facilities and Developments.* (Limited distribution to Services.)

Other reports issued by GWU. (Limited distribution to Services.)

OSRD No.	
3974	*Photographic Review of Projects Under Investigation at ABL, Review No. 1*, Aug. 1944.
	Photographic Review No. 2, Nov. 1944.
	Photographic Review No. 3, Mar. 1945.
	Contents of ABL-WPR Reports.

Report prepared under the contract, issued by the Chemical Warfare Service

The 4.2-in. Recoilless Chemical Mortar E34R1.

Reports prepared at ABL on developments there in which MIT personnel under a Division 11 contract collaborated, issued by MIT.

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MIT No.	
	Development of Portable Cordite Operated Gas Generator for Pressurizing M2-A2 Flame Thrower, A. S. Collins and A. A. Nellis, July 1945.
MIT-MR 151	*Powder Pressurizing Unit for Mechanized Flame Thrower, Preliminary Investigations*, C. H. King and C. A. Boyd, Sept. 1945.
MIT-MR 152	*Powder Pressurizing Unit for Mechanized Flame Thrower Tests with Ignited Fuels*, C. H. King, T. Q. Eliot, and R. J. Taylor, Sept. 1945.

Contract No.

OEMsr-418 California Institute of Technology

Monographs (unclassified) to be published by the McGraw-Hill Book Company.

Interior Ballistics of Rockets, Norman Wimpress.

Exterior Ballistics of Fin-Stabilized and Spin-Stabilized Rockets, Leverett Davis, Jr. and J. W. Follin, Jr.

Manuscript ready (early 1947) for publication under auspices to be arranged.

Principles of Rocket Design, W. A. Fowler and T. Lauritsen.

Final Reports, 1946 (Master print sheets of these and of this Summary Technical Report volume are to be deposited with the Coordinator of War Department Libraries, for possible photo-offset reproduction.)

OSRD No.	
2544	*Ballistic Data, Fin-Stabilized and Spin-Stabilized Rockets.*
2545	*Rocket Fuzes.*
2546	*Production of Metal Components of Rockets.*
2547	*Field Testing of Rockets: Range Operations and Metric Photography.*
2548	*Rocket Launchers for Surface Use.*
2549	*Firing of Rockets from Aircraft: Launchers, Sights, Flight Tests.*
2550	*Aircraft Torpedo Development and Water Entry Ballistics.* (Includes list of all CIT reports on this subject.)
2551	*Underwater Ballistics and Scale Models of Projectiles.* (Includes list of all CIT reports on this subject.)
2552	*Processing of Rocket Propellants.*

In the CIT numbering system, by which the reports below are listed, the first letter indicates the form or use of the report, the second the subject matter, and the third the origin (C for California on nearly all of those listed). The interim reports ("J" series) were widely distributed. Army reports in the following list which carry "A" numbers were given wide distribution under these numbers rather than under the CIT numbers. Of the periodical reports, the PMC and LMC series were widely distributed. Most of the other reports listed were given little or no distribution outside the contract activities. Most of the "J" reports were refinements and combinations of the "I" reports; the "U" series includes many ballistic tables, catalogues of rockets and components, bibliographies, abstracts, and indexes.

Prior to May 1944, OSRD numbers were assigned to only a small proportion of the CIT reports, mostly those of the "J" series. After that, all "J," LMC, and PMC reports (and a few others) bore OSRD numbers in the block 2100–2553, assigned exclusively to Contract OEMsr-418.

INTERIM REPORTS

CIT No.	NDRC No.	OSRD No.	
JAC 1	A-58	605	*Controlled Striking Angle of Rocket Projectiles*, L. Davis, Jr., and C. F. Robinson, Feb. 21, 1942. 26 pp., 3 tables, 5 ill.
JAC 2	A-115	1069	*Internal Ballistics of Jet-Propelled Devices*, B. H. Sage, Oct. 23, 1942. 46 pp., 5 tables, 27 ill.
JAC 3			*The 6-in. Rocket Motor*, J. E. Thomas, Nov. 3, 1942. 10 pp., 2 tables, 7 ill.
JAC 4	A-163	1319	*The Dependence of the Mass of Propellant in a Rocket Motor on the Web Thickness and the Motor Dimensions*, L. Davis, Jr., and Chester D. Mills, Feb. 25, 1943. 7 pp., 7 ill.
JBC 3	A-34	415	*Rocket Targets*, W. A. Fowler, Jan. 31, 1942. 35 pp., 4 tables, 11 ill.
JBC 4			*Rocket Targets*, W. A. Fowler, Apr. 14, 1942. 22 pp., 6 tables, 5 ill.
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NPC 1.44 to 1.180			*Morris Dam Weekly Reports* (Biweekly beginning Apr. 1, 1945), Aug. 23, 1942 to Aug. 4, 1945.
PMC 1.1 to 1.99, 2.1 to 2.89			*Weekly Progress Reports* (for Washington), Oct. 26, 1941 to Sept. 15, 1945. (Biweekly, beginning Apr. 1, 1945.)
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OSRD APPOINTEES[a]

DIVISION 3

DIVISION A[b]—1940-1942

Chief[c]
RICHARD C. TOLMAN

Deputy Chief
CHARLES C. LAURITSEN

Technical Aides
ELIOT B. BRADFORD
JOHN ELDER
PAUL C. FINE
HANNAH MARKSTEIN
DUANE ROLLER
EDITH TOWNES

Member
ERNEST C. WATSON

SECTION H—1940-1942

Chief[c]
CLARENCE N. HICKMAN

Vice Chairman
RALPH E. GIBSON

Technical Aides
ALEXANDER KOSSIAKOFF
PHILIP T. KIRWAN

Members
EDWIN U. CONDON
WILLIAM D. COOLIDGE
ARTHUR J. DEMPSTER
GEORGE B. KISTIAKOWSKY
ERNEST C. WATSON
WARREN WEAVER

SECTION C—1942

Chief[c]
JOHN T. TATE

Technical Aide
LORENZ G. STRAUB

DIVISION 3—1943

Chief
JOHN T. TATE
CHARLES C. LAURITSEN[d]

Technical Aides
ALVIN G. ANDERSON
ELIOT B. BRADFORD
PHILIP T. KIRWAN
ALEXANDER KOSSIAKOFF
LORENZ G. STRAUB
OTTO A. WANTUCH

[redacted]

Members

Alexander Ellett
Ralph E. Gibson
Clarence N. Hickman
Frederick L. Hovde
E. P. Hubble
George B. Kistiakowsky
William N. Lacey
Charles C. Lauritsen
Ernest C. Watson

DIVISION 3—1943-1946

Chief

Frederick L. Hovde
Eliot B. Bradford[d]

Technical Aides

Eliot B. Bradford
Frederick W. Cummings
Grace L. Hart
Roger S. Warner
Philip T. Kirwan

Members

Alexander Ellett
Ralph E. Gibson
Clarence N. Hickman
George B. Kistiakowsky
Charles C. Lauritsen
Ernest C. Watson
Louis P. Hammett

SECTON H—1943-1946

Chief

Clarence N. Hickman

Vice Chairman

Ralph E. Gibson

Technical Aides

Philip T. Kirwan
Julius A. Folse
Glenn H. Hoppin[e]

SECTION L—1943-1946

Frederick L. Hovde[d]

Technical Aide

Bayes M. Norton

[a] Many of the appointments were in effect for periods shorter than those shown for organizational units. The listing is primarily in order of appointment dates. Simultaneous appointees are arranged alphabetically.

Certain short-lived organizations are not shown above.

[b] Division A (Armor and Ordnance) included Sections C and H on rocket development, and Sections A, B, E, and T in other fields.

[c] Called Chairman through 1942.

[d] Acting Chief.

[e] Changed to Special Assistant in early 1946.

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CONTRACT NUMBERS, CONTRACTORS, AND SUBJECT OF CONTRACTS

The following list includes all contracts under which the rocket research and development programs of Division A and Division 3 were carried out. In addition, there were two purchase contracts with the Hercules Powder Company for early supplies of rocket propellant, a transfer of funds to the Army Ordnance Department for the same purpose, and transfers of 1941, 1942, and 1943 funds to the Navy Bureau of Ordnance for support of the Jet Propulsion Research Laboratory at the Naval Powder Factory, Indian Head, Maryland. Contract OEMsr-418 (which included OEMsr-250) was the only contract under Sections C and L; all others (except OEMsr-673) were related to Section H programs.

The scope of the work under each contract is indicated briefly below, and more completely by the report titles listed under each contract in Appendix Q.

Contract Number	*Name and Address of Contractor*	*Subject*
OEMsr-250	California Institute of Technology Pasadena, California	This contract included in and superseded by OEMsr-418.
OEMsr-256	Western Electric Company Bell Telephone Laboratory, Inc. New York, New York	Instrumentation for measuring rocket performance. Development of rocket launchers, of firing mechanisms, of components for Army M8 type 4.5-inch rockets, of propeller actuated ignition devices for bomb accelerators and of an electromagnetic fuze.
OEMsr-273	The George Washington University Washington, D. C. (with operations there, at Naval Powder Factory, Indian Head, Maryland, and at Allegany Ballistics Laboratory near Cumberland, Maryland)	Central laboratory operations. Ballistics research. Development and testing of instrumentation, improved propellants, flame throwers, improved mortars, recoilless mortars, and many types of rockets and related equipment.
OEMsr-416	Hercules Powder Company Wilmington, Delaware (work at Kenvil, New Jersey)	Early improvements in rocket propellants of solvent-extruded ballistite types.
OEMsr-418	California Institute of Technology Pasadena, California (with operations there at Camp Haan, at Morris Dam, at Camp Pendleton, at Naval Ordnance Test Station, all in California)	Central laboratory operations. Ballistics research. Research and development on aircraft torpedoes and other underwater ordnance. Development of instrumentation, of dry extrusion, of ballistite propellant, of all rockets, most launchers and most of the rocket fuzes used by the U. S. Navy in World War II. Pilot production of these items.

Contract Number	*Name and Address of Contractor*	*Subject*
OEMsr-671	Budd Induction Heating, Inc. Detroit, Michigan	Engineering designs and experimental production of 4″.5 rockets.
OEMsr-673	Armour Research Foundation Chicago, Illinois	Included no work on rockets. Transferred to Division 6 in late 1943.
OEMsr-702	California Institute of Technology Pasadena, California	Special studies of double base powders (work continued under Division 8 contract OSMsr-881).
OEMsr-716	University of Minnesota Minneapolis, Minnesota	Studies of the burning of double-base propellants.
OEMsr-733	Duke University Durham, North Carolina	Closed chamber studies of propellants (this contract was taken over from Division 1).
OEMsr-762	University of Wisconsin Madison, Wisconsin	Burning of double-base propellants.
OEMsr-947	Catalyst Research Corporation Baltimore, Maryland	Development of gasless delay elements for ejection charges.
OEMsr-968	Budd Wheel Company, Inc. Detroit, Michigan	Engineering design and experimental production of metal components for many rockets and mortars.

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SERVICE PROJECT NUMBERS

The projects listed below were transmitted to the Executive Secretary, National Defense Research Committee, NDRC, from the War or Navy Department through either the War Department Liaison Officer for NDRC or the Office of Research and Inventions (formerly the Coordinator of Research and Development), Navy Department.

Service Project Number	*Subject*
AC-52	Development of a specially shaped bomb (referred to as a water plunge bomb) designed to follow a horizontal path in water after being dropped at high speed from aircraft.
AC-70	Hydrobomb (torpedo for Army aircraft).
AC-121	Development of sights for firing aircraft rockets.
CWS-10	Development of flame throwers (including their pressuring by propellant gases).
CWS-22	Rocket projection of chemical munitions (and extension of range of 4″.2 chemical mortar).
CWS-30	Development of 4″.2 recoilless mortar and shell.
CWS-34	Improvement of 4″.2 chemical mortar.
NA-167	Study of nozzle design for jet motors.
NA-197	Development of jet-assisted take-off unit for carrier based aircraft.
NA-231	Assistance on the development of aircraft launching equipment.
NO-33	Internal and external ballistics of rockets; and double-base propellants for rockets.
NO-34.1	Rockets for aircraft armament.
NO-34.2	Low altitude antiaircraft rockets.
NO-34.3	High altitude antiaircraft rockets.
NO-35.1	Jet accelerators for armor-piercing bombs.
NO-35.2	Rockets for assisted airplane take-off (including rocket catapult).
NO-36.5	Rocket projection of antisubmarine depth bombs from ships.
NO-39.1	Rocket targets.
NO-99	Jet propulsion (solventless extrusion at Bruceton).
NO-116	Scatter bombs for attack of submarines by airplanes.
NO-118	Rocket weapons (for beach barrage in amphibious assault).
NO-120	Parachute rocket flare (for identification of warfare targets from aircraft).
NO-121	Retro-rocket bombs (for attack of submarines by MAD-equipped airplanes).
NO-140	Horn-type retro-bombing fuze.
NO-141	Hydrodynamic characteristics of projectile forms.
NO-146	Underwater trajectories of depth charges.
NO-148	Torpedo launching mechanism (design and construction of).
NO-153	Smoke float rocket and projector, development of.
NO-164	3″.25 rocket and projector.
NO-165	Rocket projectors (for Marine Corps, development of, and establishment of test ranges).
NO-168	Rocket deceleration of aircraft launched torpedoes.

Service Project Number	*Subject*
NO-170	Adaptation of Navy 3″.25 and 5″.0 rockets to aircraft (and development of 5″.0 high velocity aircraft rockets).
NO-176	Torpedoes for high speed aircraft (including water entry tests).
NO-177	Jet-propulsion of aircraft torpedoes.
NO-192	Shipboard rocket launcher for the 3″.25 rocket, development of.
NO-196	Anti-surface vessel ordnance.
NO-204	Contact fuzes, development of (for depth bombs).
NO-205	Rocket targets (production of).
NO-214	Ballistic range converter for ASD radar (for forward firing aircraft rockets).
NO-215	Spin stabilized rockets.
NO-216	Aircraft sight for forward firing rockets of CIT 3A type.
NO-227	Subcaliber training rocket for aircraft use.
NO-228	Design, construction, and operation of extrusion presses (at Naval Ordnance Test Station, Inyokern, Calif.).
NO-230	3″.25 window rockets.
NO-238	Development of launchers for spin stabilized rockets.
NO-245	Development of high performance 4″.5 aircraft rocket.
NO-246	Development of 5″.0 aircraft rocket.
NO-247	Development of 2″.36 high velocity rocket, H.E.A.T.
NO-248	Development of improved components for 4″.5 rocket, M8 type.
NO-249	Development of spin-stabilized rocket using solvent-extruded propellants.
NO-250	Rocket projection of bombs.
NO-251	Development of 3″.25 or 3″.5 rocket.
NO-252	Development of 3″.25 rocket, multiple grain, thin web.
NO-253	Development of 7″.2 rocket motor.
NO-254	Development of 10″ rocket motor.
NO-256	Forward firing large caliber aircraft rockets ("Tiny Tim").
NO-259	Demolition rockets and launchers.
NO-260	Scoring of air to air rocket firing.
NO-271	Experimental production of 3″.25 rocket motors, Mk 5, for CWR-N rockets.
NO-280	Statistical assistance on the analysis of firing data for rocket propellant.
NO-282	Development of 2,000 lb forward firing, large caliber aircraft rocket.
NO-284	Development of aircraft rocket sights.
NO-289	Assistance to the Naval Ordnance Test Station, Inyokern, Calif.
NO-296	Development and fabrication of launching rockets for Bumblebee.
NS-164	Rocket propulsion to insure proper ejection of Mk 2 grenade from new airless emergency signal ejector of submarine.
NS-211	Countermeasure for antisubmarine contact fuzed charges.
NS-309	3″.0 solid slow burning propellants (for generating gases to drive turbine).
OD-14	Special fuels for jet propulsion and squib igniter performance.
OD-26	Jet propulsion (development of many early rockets; superseded by OD-161 to 172).
OD-66	Device to determine direction and range of a forward artillery officer from immediate vicinity of a battery position.
OD-98	Rocket targets (with wings).
OD-125	Long range (75 miles) rocket projectile.

SERVICE PROJECT NUMBERS (*Continued*)

Service Project Number	*Subject*
OD-137	Demolition rockets and launchers.
OD-155	Factors which control afterburning in rockets.
OD-161	Development of high performance 4″.5 aircraft rocket.
OD-162	Development of 5″ aircraft rocket.
OD-163	Development of 2″.36 high velocity rocket H.E.A.T. (and of electromagnetic fuze for it).
OD-164	Development of 3″.25 rocket, single grain, solventless powder type.
OD-165	Development of improved components for 4″.5 rocket, Mk 8 type.
OD-166	Development of spin-stabilized rockets using solvent extruded propellant.
OD-167	Development of spin-stabilized rockets using solventless-extruded propellant.
OD-168	Rocket projection of bombs (by standard rocket motors).
OD-169	Development of 3″.25 or 3″.5 rocket.
OD-170	Development of 3″.25 rocket, multiple grain, thin web.
OD-171	Development of 7″.2 rocket motor.
OD-172	Development of 10″ rocket motor.
OD-179	Statistical assistance on the analysis of firing data for rocket propellant.
OD-183	Bourdon systems (for measurement of performance of rocket propellant grains).
OD-184	Development of powder charge assembly for recoilless mortar, 60 mm.
OD-185	Development of stationary rocket motor, 3 to 3.5 inch, for special H.E.A.T. projectile (includes 81 mm recoilless mortar).
OD-186	Minefield clearing devices of the jet-propelled type.
OD-187	Adaptation of Tiny Tim rocket motor.
OD-196	Multiple-cartridge, tube-launching system for JB-2.
OD-199	Rocket accessories for aircraft.
OD-201	Research on elements of rocket motors with high impulse ratio.

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